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Volume III

DURABILITY METHODS DEVELOPMENT
VOLUME III
STRUCTURAL DURABILITY SURVEY: STATE-OF-THE-ART ASSESSMENT

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This technical report has been reviewed and is approved for publication.

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

A structural state-of-the-art assessment was conducted to evaluate the durability of previous and existing aircraft structures. The assessment includes data obtained from literature, previous and current programs, and visits to all Air Force Air Logistics Centers. Results for each major Air Force aircraft type are summarized showing trends in structural and material degradation. The most frequently occurring in-service structural problems, in order of occurrence, were found to be cracking, corrosion and fastener-related problems.

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F O R E W O R D

This program is conducted by General Dynamics, Fort Worth Division with George Washington University (Dr. J. N. Yang) and Modern Analysis Inc. (Dr. M. Shinozuka) as associate investigators. This program is being conducted in three phases with a total duration of 50 months.

This report was prepared under Air Force Contract F33615-77-C-3123, "Durability Methods Development". The program is sponsored by the Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, with James L. Rudd as the Air Force Project Engineer. Dr. B.G.W. Yee of the General Dynamics' Materials Research Laboratory is the Program Manager and Dr. S. D. Manning is the Principal Investigator. This is Phase I of a three phase program.

This report (Volume III) is one of the five volumes that documents the Phase I work. The other Phase I reports are:

Volume I	-	Phase I Summary
Volume II	-	Durability Analysis: State-of-the-Art Assessment
Volume IV	-	Initial Quality Representation
Volume V	-	Durability Analysis Methodology Development

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SUMMARY

The objective of the Durability Methods Development program is to develop and verify a durability analysis methodology for satisfying the Air Force's durability design requirements for metallic airframes. A structural state-of-the-art (S.O.A.) assessment was initially performed to determine the most frequently occurring forms of structural durability degradation for in-service Air Force aircraft. This assessment was essential to assure the durability Analysis Methodology developed would apply to the most prevalent form of structural durability degradation observed for in-service aircraft. This Volume (III) documents the results of a comprehensive structural degradation survey of Air Force aircraft.

The structural S.O.A. assessment was performed in three stages: data acquisition, data analysis and documentation. The following Air Force Air Logistic Centers (ALC) were visited to gather in-service information at the maintenance depot level:

o San Antonio ALC	Kelly AFB, Texas
o Oklahoma City ALC	Tinker AFB, Oklahoma
o Warner - Robins ALC	Robins AFB, Georgia
o Sacramento ALC	McClellan AFB, California
o Ogden ALC	Hill AFB, Utah

Data gathered at these ALC's included written reports, i.e., Aircraft Structural Integrity Plans (ASIP), Analytical Condition Inspection (ACI), fatigue test teardown results, etc., and direct contacts with ALC personnel. Valuable information was also obtained from various aircraft company reports and General Dynamics, Fort Worth Division data.

The following aircraft systems were included in the structural survey:

- o Fighter Aircraft
  - F-100
  - F-104
  - F-105
  - F/RF-100
  - F-106
  - F-4C/D/E
  - F-111
  - F-15
  - F-16

- o Trainer Aircraft
  - T-37B/C
  - T-38
  - T-39
- o Bomber Aircraft
  - B-52
  - FB-111
  - F-111C
- o Cargo/Transport
  - C-130
  - C-141A
  - KC-135
  - C-5A
- o Attack Aircraft
  - A-7
  - A-10

Results of these surveys are reported herein in various forms.

The most frequently occurring in-service structural problems, in order of occurrence, were found to be cracking, corrosion and fastener-related problems. A uniform format is needed for documenting durability-related problems at the Air Logistics Centers. Data should be compiled in useful formats for storage and retrieval and should be periodically updated.

SECTION I

INTRODUCTION

The durability state-of-the-art assessment task is the initial effort in the Durability Methods Development Program for which the overall objective is to establish the durability procedures and design techniques necessary to minimize fatigue cracking and/or other structural material degradation in advanced metallic aircraft structures.

The durability state-of-the-art assessment (S.O.A.) task defined by this program is a two-fold task, namely, to conduct:

1. Structural State-of-the-Art Assessment

2. Analytical State-of-the-Art Assessment

This report presents the results of the structural state-of-the-art assessment. The analytical effort is presented in Reference 1 .

SECTION II

STRUCTURAL STATE-OF-THE-ART ASSESSMENT

2.1 STRUCTURAL S.O.A. ASSESSMENT APPROACH

The overall S.O.A. assessment approach was to conduct an extensive assessment of the durability of previous and existing aircraft structures. This assessment includes information obtained from literature surveys, previous and current programs, contractor and subcontractor sources,and personal visits to the five Air Force Air Logistic Centers.

The overall program plan for the structural S.O.A. assessment is shown in Figure 1.

2.1.1 Structural S.O.A. Assessment Objectives

The primary objective of the structural assessments as detailed in Appendix A, "Durability Methods Development Structural Assessment Results", was to determine the following:

1. Types of aircraft structure considered
2. Forms of structural and/or material degradation
3. Aircraft location
4. Cracking failure description
5. Probable flaw type/cause other than fatigue or corrosion

2.1.2 Forms for Documentation of Structural and/or Material Degradation

During the early planning of the structural assessment task, an idealized form was developed that would encompass the various forms of structural and/or material degradation experienced on aircraft structures (see Figure 2). The primary purpose of the form was to provide an in-depth insight into the different

OVERALL PROGRAM PLAN

- o AIRCRAFT TYPES
- o DURABILITY STRUCTURAL ASSESSMENT OBJECTIVES
- o CONDUCT DURABILITY DATA SEARCH & ACQUISITION
- o CONDUCT STRUCTURAL STATE-OF-THE-ART ASSESSMENT

FIGURE 1 Structural State-of-the-Art Assessment Program Plan

DURABILITY STRUCTURAL ASSESSMENT OBJECTIVES

(1) FORMS OF STRUCTURAL AND/OR MATERIAL DEGRADATION

- o Fatigue
- o Corrosion
- o Wear
- o Fretting
- o Galling
- o Other
- o Frequency of Occurence
- o Severity
- o Etc.

(2) AIRCRAFT LOCATION

- o Wing
- o Fuselage
- o Horizontal Tail
- o Vertical Tail
- o Other

(3) CRACKING FAILURE DESCRIPTION

- o Size
- o Shape
- o Location
- o Quality

(4) PROBABLE FLAW TYPE/CAUSE OTHER THAN FATIGUE OR CORROSION

- o Manufacturing
- o Material Imperfection
- o Installation
- o Quality Control
- o Improper Maintenance
- o Repair Related
- o Design Deficiency
- o Material Selection

(5) FLEET IMPACT

- o None
- o Minor Repair
- o Major Mod

FIGURE 2 Durability Survey Outline

forms of durability which occur on aircraft structure, their frequency of occurrence and severity. The form also provided for an assessment of the initial quality, original design practices applied, types of manufacturing flaws/defects present or induced, cause and criticality of such flaws, etc. This form, when completed, could in itself reveal the types of damage which occur in aircraft structures, the relationships between these types of damage and the initial quality of the structures, and the parameters which contribute to the initial quality of the structures.

During the Air Logistic Center visits, it became apparent very early that any attempt to obtain all the in-depth information desired by the assessment form was fruitless. Rationale for this conclusion was obvious; namely,

1. No systematic form of record keeping for such data exists.

2. No prior requirement for documenting these types of data.

3. Concern more for repair, fix, mod rather than cause-effect relationship

4. Budget-manpower restraints.

Although the structural S.O.A. assessment detail form was abandoned for the ALC visits, a significant durability data base was obtained for the various aircraft types from the Air Logistic Centers. These data are summarized and presented in Section III .

2.2 DATA SEARCH AND ACQUISITION

The data search and acquisition for the structural assessment task was almost unlimited as shown in Figure 4. Perhaps the most significant source for data was the visits to all five Air Force Air Logistic Centers. Data obtained from these centers provided a significant data base for the structural S.O.A. assessment task. These data along with those from other sources are presented in Section IV of this report.

Another data source used in this task was the literature survey. Data obtained via this method include symposium papers, periodicals, reports, etc. Still another very significant data source was the abundance of in-house data generated from both previous and current component/element test and full scale test programs such as those listed in Figure 3 . Other data sources include results from teardown inspections, analytical condition inspections, and miscellaneous CRAD and IRAD programs.

2.2.1 Air Logistic Center Visits

Probably the most significant input to the structural S.O.A. assessment task was the first-hand information gathered from the Air Force Air Logistics Centers. During this task visits were conducted to each of the five ALCs with a repeat visit to San Antonio. A list of the centers visited, dates, and the aircraft for which durability data were obtained is shown in Figure 4. Data gathered from the ALCs for the Task I Structural S.O.A. assessment are discussed and presented in Section III.

2.2.2 Full Scale Test Programs

Durability full scale test program results were obtained during Task I for practically every airframe type surveyed. These include fighters, bombers, trainers and cargo/transports. These data are discussed and presented in Section III.

2.2.3 Component Element Test Programs

Durability test results from various component/element tests were also included in the structural S.O.A. assessment task. Among the programs discussed and presented in Section III are the following:

- F-111A (FW #1) Wing Carry-Thru Box Component
- F-111A (FW #2) Wing Carry-Thru Box Component
- F-111A (A-4) Vertical Tail Component
- F-111A (A-4) L/H Wing Test Component
- ECP 10212 - F-111A Wing Test Component
- AMAVS - Advanced Metallic Air Vehicle Structures Test
- ECP 613 -F-4B Wing Test Component
- Fastener Hole Quality
- Other

2.2.4 Other

Another source from which durability data was obtained for Task I was inspection programs. These data include results from Analytical Condition Inspections (ACI), Special Inspections, Periodic Depot Maintenance (PDM) Inspections, and Teardown Inspections.

CRAD and IRAD programs are among the "Other" data surveyed during Task I. These data are also discussed and presented in Section III.

2.2.5 Aircraft Types Surveyed

An attempt was made to survey as many Air Force type aircraft as possible for the Task I assessment and yet stay within the program schedule and budget. The source of durability data that could be gathered for all types of aircraft operating under the many usage and environment criteria is almost unlimited and certainly beyond the scope of this task. However, the survey conducted did include a very good representation of the Air Force fleet which includes fighters, bombers, trainers, cargo/transports, reconnaissance and attack aircraft. A summary of the aircraft surveyed for which data are presented in Section III is shown in Figure 5.

DURABILITY DATA SEARCH AND ACQUISITION FROM :

SERVICE AIRCRAFT HISTORY SURVEY

- o Literature
- o Base Visits
- o Contractor Sources
- o Accumulated In-House Data
- o Previous and Current Flight Test Programs
- o Other

COMPONENT/ELEMENT TEST PROGRAMS

- o F-111 Wing Carry Thru Box (CTB)
- o FW No. 1-3
- o FW No. 1-4
- o F-111 Wings (L&R)
- o ECP 10212 Wing (FB-111)
- o F-16 A/B Wing Fuselage Dam. Tol. (2)
- o Fastener Hole Quality
- o Other (Coupon Specimens)
- o Advanced Metallic Air Vehicle Structure Test (AMAVS)

FULL SCALE TEST PROGRAMS

- o F-111A Cyclic Fatigue Test
- o FB-111A Cyclic Fatigue Test
- o F-16 Durability & Damage Tolerance Test
- o F-4
- o A-7
- o C-130
- o Others

OTHER SOURCES

- o Analytical Condition Inspections (ACI)
- o F/FB-111
- o F-16
- o Miscellaneous Programs

FIGURE 3 Durability Survey Sources

DURABILITY DATA SEARCH AND ACQUISITION FROM FLEET AIRCRAFT

ALC VISITS	DATES	AIRCRAFT SURVEYED
o SA-ALC - San Antonio, Tx.	25 May 78 17/18 Aug. 78	T-37, T-38, F-106 C-5
o WR-ALC - Robins, Ga.	11/12 July 78	C-130, C-141, F-15
o OC-ALC - Oklahoma City, Okla.	18/19 July 78	B-52, KC/135, A-7D
o SM-ALC - Sacramento, Calif.	25/26 July 78	F/FB-111, T-39, F-100, F-104, F-105, A-10
o OO-ALC - Ogden, Utah	27/28 July 78	F-101, F/RF-4

FIGURE 4 Air Logistic Centers Visited

AIRCRAFT TYPES**

FIGHTERS

- o F-100
- o F-101
- o F-104
- o F-105
- o F-106
- o F-111
- o F-4
- o F-15A
- o F-16A/B

TRAINERS

- o T-38
- o T-37
- o T-39

BOMBERS

- o B-52
- o FB-111A
- o F-111C

CARGO/TRANSPORTS

- o C-130
- o HC-130
- o KC-135
- o C-141A
- o C-5A

RECONNAISSANCE

- o RF-4C

ATTACK

- o A-7D
- o A-10

OTHER

- o Observation
- o Patrol
- o Early-Warning
- o Utility - T-39

**Significant durability data has been retrieved for all the above aircraft during the ALC visits.

FIGURE 5 Aircraft Types Surveyed

SECTION III

DATA RESULTS

3.1 FIGHTER AIRCRAFT

3.1.1 F-100 Series

The F-100A is a single-place, low-wing air superiority fighter with level flight capability in excess of Mach 1.3.

The F-100C represents an air superiority fighter incorporating special provisions for fighter-bomber capabilities.

The F-100D aircraft, essentially similar to the C model, is equipped with autopilot and radar.

The F-100F has a two-place tandem cockpit which gives it trainer as well as fighter and fighter/bomber capability.

Durability problems experienced with this aircraft are:

- WING CENTER SECTION LOWER SKIN
  - Fatigue cracks in 5/16" dia. hole at mid-chord
  - Numerous fatigue cracks at fastener holes revealed by fleet inspection.
- WING OUTER PANEL LOWER SKIN
  - Fatigue cracking in fastener holes during test.
  - Two service crashes in same area.
  - Added doubler as interim fix for fleet mod.
  - T-Bird fillet radius cracks @ WS 33.8
- WING OUTER PANEL TO WING CENTER SECTION JOINT FATIGUE
  - Fatigue in outboard row of bolt holes.

- o WING ROOT RIB
  - Stress corrosion caused by press-fit bushings
  - Stress corrosion in wing spar cap sealant grooves.
- o INDUCED FATIGUE IN PRIMARY STRUCTURAL MEMBERS CAUSED BY FASTENER INSTALLATION
  - Deutsch Fastener Installation
  - Rozan Insert Installation

3.1.2 F-104 Series

There are five basic models of the F-104 aircraft from which nineteen different configurations are derived.

The F-104A is a single-place, supersonic type fighter aircraft with a principal mission of interception and destruction of enemy aircraft.

The F-104B is a two place, supersonic fighter aircraft. Its principal mission is interception and destruction of enemy aircraft and to serve as a proficiency and transitional trainer.

The F-104C is a single-place supersonic-type interceptor for aircraft. Its principal mission is the interception and destruction of enemy airborne weapons during day or night.

The F-104D is a two-place, supersonic-type aircraft. Its principal missions are interception and destruction of hostile aircraft under "air superiority fighter conditions" and to serve as a proficiency and transitional trainer.

The F/RF-104G is a single place, supersonic-type aircraft. Its missions are all-weather delivery of tactical weapons, all-weather interception and destruction of enemy aircraft and photo reconnaissance.

The more significant durability problems encountered with this aircraft are:

- o LOWER WING ATTACH FITTINGS NOS. 1 & 5
  - Fatigue cracking in aft fastener hole - 7079-T6 and 7075-T6 materials

- o LOWER WING SKIN @ WS 80.7 AILERON SERVO ACCESS OPENING

  - Fatigue cracking in access opening - 7075-T6 and T-651 materials

- o LOWER WING SKIN @ WS 47

  - Minute fatigue cracking in bore of fastener holes.

3.1.3 F-105 Series

There were five different models of the F-105 produced of which three remain in active inventory, namely, F-105B/D/F aircraft.

The F-105B is a single-seat, supersonic all weather fighter-bomber with nuclear capability.

The F-105D is basically the same as the F-105B with updated avionics.

The F-105F is the same as the F-105D except it has a second cockpit in tandem with dual controls for trainer capability.

Durability problems experienced with this aircraft are:

- o FRONT SPAR

  - Stress corrosion cracking in 200-250 A/C caused by shim problem in flange to web radius of 7075-T6 material.

- o REAR SPAR

  - Stress corrosion cracking in forward web of 7075-T6 forging

- o MLG PORK CHOP FITTING

  - Fatigue cracking in large radius. Steel fitting replaced aluminum fitting for production.

- o HORIZONTAL STABILIZER
  - Stress corrosion cracking in 7079-T6 crossbeam forging.
- o WING INBOARD PYLON FITTING
  - Corrosion fatigue in 80% of fleet
- o FUSELAGE STATION 390 FRAME
  - Fatigue cracking accelerated by corrosion in 7075-T6 material
- o FUSELAGE STATION 442 FRAME
  - Cracking in undercut lugs of 4330V steel forging.
- o UPPER FUSELAGE SKIN SPLICES, SPOT WELDS AND FUEL CELL ACCESS OPENINGS
  - Manufacturing defects in countersunk fastener holes at fuel cell access opening caused fatigue cracking.
- o POOR MANUFACTURING PROCEDURES, POOR INSTALLATION AND ROUGH FIELD HANDLING
  - Manufacturing -- no radius in base of C bore
  - Rough field handling -- tools banged against structure with disregard for dents, scratches, notches, etc.
  - Poor installation -- high torque required in torquing 3/4" bolt in wing/fuselage attach fitting. No regard for handling of tools in relation to structure.
- o TRAILING EDGE FLAP SUPPORT
  - Fatigue initiated from corrosion pit caused by metal contact between dissimilar materials, 7075-T6 forging and steel roller track.
- o WING LUG POCKETS AND WHEEL WELL
  - Stress corrosion in 15 fleet aircraft.

3.1.4 F/RF-101 Aircraft

These aircraft are charged with the primary role of air defense and tactical reconnaissance for both coastal and inland environment.

Primary durability problems experienced with the early aircraft prior to fixes were:

- o LOWER WING SKIN CRACKING
  - Fatigue cracking in fastener holes
- o MAIN SPAR CARRY THRU STRUCTURE
  - Diagonal cracking

Analytical condition inspections were performed in CY 1976 and CY 1977 on ten of these aircraft. The results are as follows:

- o F/RF-101 SUMMARY OF ACI RESULTS

DEFECTS	CY 1976	CY 1977
Minor	12	60
Major	1	4
Critical	0	0

Minor defects consisted primarily of loose/missing fasteners, mild corrosion in wing fuel cell cavities, deteriorated paint and sealant, cracked flair in fuel line, etc.

Major defects consisted of advanced corrosion and delaminated fuel cell.

3.1.5 F-106 Series

The F-106A is a single-place, supersonic all-weather interceptor recently modified to include Air Combat Tactics.

The F-106B is a two-seated version of the F-106A.

Potential durability problems associated with these aircraft based on the full scale fatigue test to 36,000 hours (scatter factor = 4) are as follows:

- LOWER WING SKIN @ ELEVON ACTUATOR CUTOUTS
  - Fatigue cracking in inboard and outboard corner radii.
- WING SPAR
  - Cracking in lower lug bolt hole of Spar 2
- FUSELAGE UPPER LONGERON
  - Fatigue cracking on inboard leg at the cockpit aft pressure bulkhead
- FUSELAGE BELT FRAMES
  - Cracking of belt frames due to excessive stiffness in wing drag angle.

No major structural failures have been reported from analytical condition inspections.

3.1.6 F-4 Series

The F-4C/D and E aircraft are supersonic, all-weather, tactical fighter aircraft with special weapons capability.

The F-4D is equipped with additional fire control and expanded weapons capability.

The F-4E has engines designed for increased acceleration capability. The leading edge at the stabilator is slotted for increased control at lower landing speeds, and a wing modification provides leading edge slats to enhance air-to-air combat capability.

The RF-4C is a tactical reconnaissance aircraft capable of high-low, day-night selected reconnaissance missions.

Durability problems experienced on these aircraft are quite representative of the fighter aircraft series. Among these problems are the following:

- WING SKINS
  - Fatigue is the primary damage source
  - Main structural skin failures originating at fastener holes
- TORQUE BOX DAMAGE
  - Gouging caused by maintenance
- TORQUE BOX SKIN
  - Fatigue cracking in 1.86 inch pylon access hole
  - Torque box skin fastener holes have been trouble free due to installation of taper lok fasteners and cold working of holes.
- CENTERLINE RIB
  - Multiple cracking in lower flange at fastener holes caused by stress corrosion
  - Some fatigue in bolt holes and radii
- MAIN SPAR
  - General cracking
- FRONT SPAR
  - Usually trouble free
- OUTER WING
  - Skin cracking caused by fatigue in fastener holes
  - One airplane lost due to catastrophic failure
  - Hole elongation in skin due to compression of skin caused by buffet loads not accounted for in analysis.

- o FUSELAGE - STABILATOR ACTUATOR HORN
  - Stress corrosion in forging flashing line
  - Navy lost an aircraft from this problem
  - Material change from 7075-T6 to 7075-T773
- o TURTLE BACK DOORS
  - Hole elongations from wear and door deflections
- o FUEL CELL CAVITY
  - 100% fatigue cracking in 14-inch curve span
- o FLOOR AREA
  - Fatigue cracking in thin metal floors caused by unconservative analysis of slosh loads and fuel pressure
- o FUSELAGE LONGERONS
  - Generally good except cracking found on longeron flange caused by maintenance using flange for a step.
- o GENERAL OBSERVATIONS
  - Most cracking showed up downstream of predictions; however, some fastener hole cracking in sealant grooves appeared much sooner.
  - One fatigue area (most aft lug on inboard wing forward rib) surfaced in service that did not show in test. Cause of failure was a very sharp radius.
  - Cockpit blanket insulation provides a trap for moisture condensate on inner skin resulting in corrosion of inner skin.

Significant test and service failures are summarized in Figures 6 and 7, respectively. Service failures attributed to stress corrosion are shown in Figure 8.

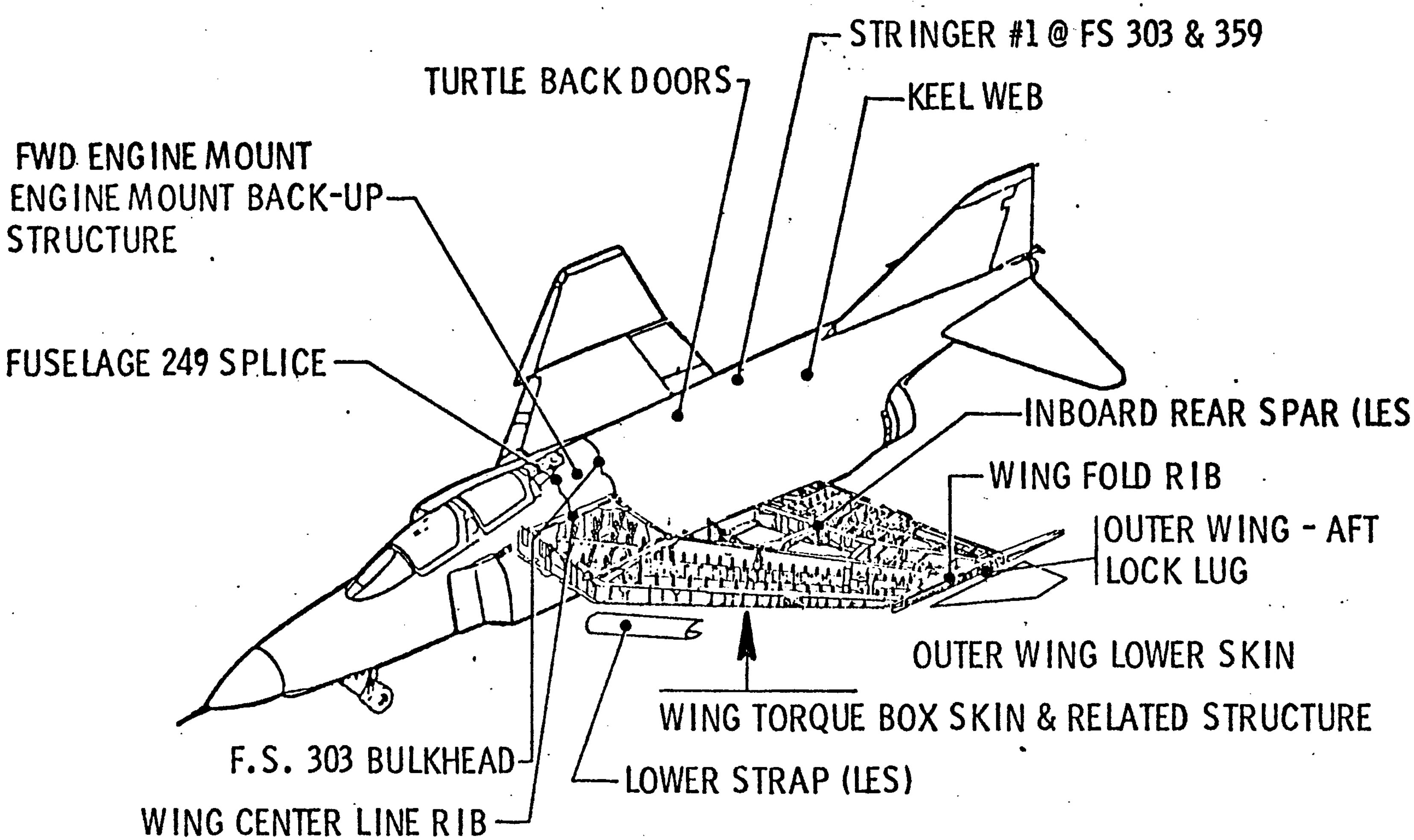


FIGURE 6 F-4 Significant Test Failures

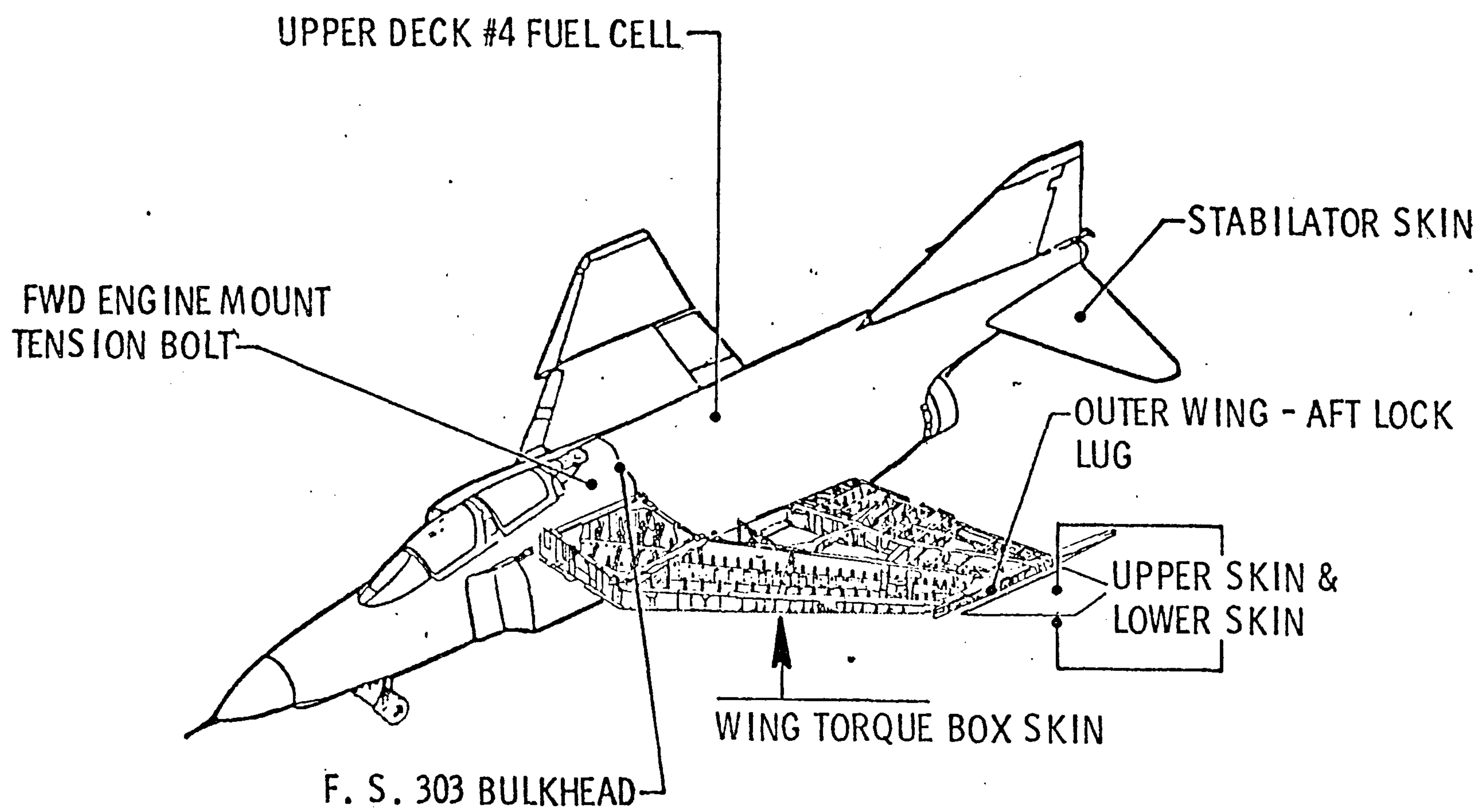


FIGURE 7 F-4 Significant Service Failures

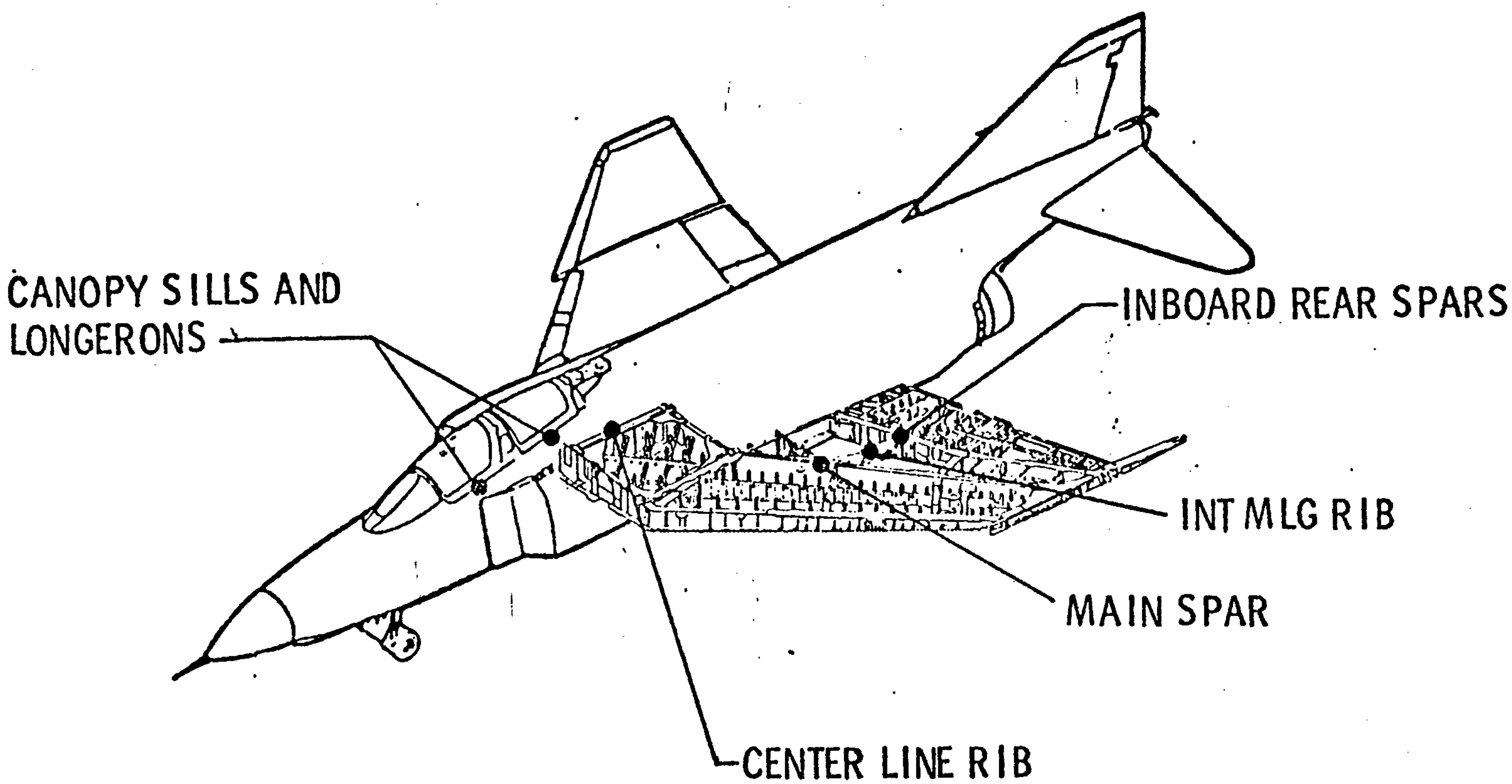


FIGURE 8 F-4 Significant Service Failures
(Stress Corrosion)

3.1.7 F-111 Series
(Ref. 2 : F/FB-111 Aircraft Structural Integrity Program

The F-111A is a two-place side-by-side Mach 2 plus, fighter type aircraft powered by two TF 30-P-3 engines and is configured with a variable sweep wing (for short take-off/landing capability).

The F-111E aircraft are the same as the F-111A except for the inlet geometry which is of the Triple Plow II configuration.

The F-111D aircraft are equipped with Mark II avionics and are powered by TF 30-P-100 engines and TP II inlet geometry.

The F-111F aircraft have the TF 30-P-100 engines, TP II geometry, an improved wing carry-thru box, Mark II avionics and increased strength main landing gear as changes from the F-111A configuration.

Due to the significant contribution the F-111 made to the field of structural integrity by being the fore-runner in the development and application of fracture mechanics technology to aircraft structures, it is felt that a background of this information is appropriate. The following data has been extracted from the Ref. F-111 ASIP Master Plan Document.

F-111 ASIP Master Plan Document

2.0 SUMMARY

The design, development and certification of the F-111 structure has followed the principles defined in ASD TR 66-57 "Air Force Aircraft Structural Integrity Program: Airplane Requirements". The basic requirements in terms of aircraft type, mission, required service life, and other performance parameters which establish the structural design criteria for both static strength and fatigue life were specified in the original request for proposals in 1961.

2.1 Importance ASIP Development

2.1.1 The demanding performance of modern aircraft in the areas of speed, altitude, and range places a premium on minimum weight structural design. In the case of the F-111, the constraints to provide minimum structural weight design within a minimum structural

envelope, and provide for variable wing sweep were met by the monolithic wing pivot and wing carry-thru box structures fabricated as D6ac steel weldments. Also, to further the minimum weight design concept, the steel was subjected to a higher heat treatment to obtain high strength capability.

In December 1969, F-111A Nr. 94 (S/N 67-049)crashed as a result of a wing failure in the wing pivot fitting. The failure was caused by an undetected flaw in the wing pivot fitting forging at the time of fabrication.

Initial in-process inspection and non-destructive test capability did not provide for the sensitivity of high strength materials to small flaws and defects which were acceptable for past alumimum/steel designs and fabrication practices. This problem of pre-existent flaws was separate and distinct from the classic strength and fatigue problems identified by the structural integrity programs defined by ASD TR 66-57 and its predecessors. For this reason, the F-111 ASIP and the Air Force ASIP added fracture mechanics procedures for design, structural analysis, manufacturing methods, process controls, and inspections.

2.1.2 Fatigue test failures in the F-111A carry-thru box structure in August 1969, February 1969, and June 1969 revealed the need to design fixes and construct new test specimens. The resulting tests also incorporated a rational and realistic test spectrum derived from projected operational usage in order to facilitate interpretation of test results in terms of programmed aircraft fleet life. Due to changes in test spectra, the F-111 contract requirements for a 4000 hour service life (16,000 test hours) was interpreted to be met by fatigue testing to a damage level which results in a safeservice life of 6,000 hours for the tactical fleet (24,000 test hours). The FB-111A was tested to demonstrate a safe service life of 10,000 hours (40,000 test hours) based on a typical bomber spectrum. An improved safe life wing carry-thru box was designed with significantly lower working stress levels and with specific design changes to account for the critical areas defined by previous

tests. This improved WCTB is being incorporated in the F-111F and has been retrofitted into the F-111C. Tests have shown that high heat treat steel is extremely sensitive to surface finish especially in areas of stress concentrations. For this reason, field operations must pay careful attention to protection of the surface finish of the primary steel structure from mechanical damage and corrosion.

2.1.3 The Phase 1 "Recovery Program" was established in February 1970 following the crash of F-111A 67-049 to restore fleet aircraft operation to 80% of design capability. The sensitivity of high strength materials to small flaws (e.g., the wing pivot fitting on aircraft 67-049)and to surface finish in areas of stress concentrations (i.e., fatigue test results 1968-69) were the primary factors in determining the short term actions required for the program.

Included in the program was the proof test loading of each aircraft to design limit load conditions (+7.33g and -2.4g) at a wing sweep of 56^{o} and a temperature of -40^{o}F. Cold test chambers were constructed at both the contractors facility (Fort Worth, Texas) and at Sacramento ALC. The proof test program was completed in February 1972 on 340 F-111A/D/E and FB-111A fleet aircraft. Cold chamber proof testing was also included in acceptance testing of the remaining production aircraft, and was a part of the F-111C modification program.

Since the cold proof test was to demonstrate that fleet aircraft were free of significant structural discrepancies in the high strength steels, the two failures encountered prevented the affected aircraft from crashing (with probable fatalities) upon return to the fleet. F-111A Nr. 43 (S/N 66-025) and F-111E Nr. 75 (S/N 68-065) failed in the wing carry-thru box (WCTB) and horizontal tail pivot fitting, respectively. Both failures had causes associated with the sensitivity of the material to fabrication and processing.

A second phase of the program (categorically a depot level maintenance task) was called "Phase II Structural Inspection Program (II SIP) and included

incorporation of structural modifications required to provide full (100%) strength and service life design capability in the fleet aircraft. Updated NDI techniques were used during modification and inspection, and during the second cold proof test (a II SIP requirement). The cold proof chamber at Sacramento ALC was modified for the second test to incorporate a 26° wing sweep, positive design load test condition at -40° in addition to the original 56° wing sweep tests at -40°F.

The static test and fatigue test programs were successful in the context that minimal weight increases and few changes for structural strength have resulted from normal development testing. The significant aspects of the fatigue test program were noted in Paragraph 2.1.2 above.

Both the static and fatigue test articles were initially configured as full-scale total airframes. In November 1968, the structural integrity test program was realigned to reduce completion time of the structural certification effort. This decision was, in part, motivated by the catastrophic failure of the wing carry-thru box (August 1968) in the fatigue test article, and the resulting loss of the fatigue test article for continuation of testing. Both test articles were reconfigured into major component full-scale articles consisting of fuselage/empennage, wing, and horizontal tail. Future failures in any component would not delay the total programs.

Phase 1, Design Information

The basic structural design information for the F/FB-111 MDS is delineated in the documents presented in Section 7.0, Part 7.1.

<u>ASIP Requirements Document and Applicable Military Specifications</u>

The requirements for the F-111 ASIP were defined in ASD-TN-61-141, dated September 1961, which was incorporated into the F-111 contractual specifications as

an applicable document. ASD-TR-66-57 (the updated version of TN-61-141) was used as the basis for accomplishment of the F-111 ASIP. The applicable baseline military specifications were the MIL-A-8860 (ASG) series specifications, dated 18 May 1960, and MIL-S-5711 (USAF), dated 14 December 1954. The MIL SPECS were amplified and modified for the F/FB-111 MDS by appropriate contractual documents (e.g., structural design criteria reports).

Basic Strength Requirements

Table 3.1.1 is a compilation of these requirements in terms of load factors and the associated design gross weights.

Basic Fatigue Requirements

These requirements are shown in Table 3.1.2 in terms of flight hours and landings. The design mission profiles and mix for the F-111A/D/E/F and FB-111A are shown in Section 6.0, Appendix C-1.

Basic Flutter, Vibration, and Acoustic Requirements

These requirements were defined in SPEC MIL-A-8870 (ASG) dated 18 May 1960.

Guaranteed Weight Empty

None of the F/FB-111 MDS was designed to a guaranteed weight empty. However, all contractor responsible weight increases were required to be included for purposes of determining strength compliance.

Changes to Initial Requirements

The fatigue spectrum was modified from the MIL-A-886 (ASG), dated 18 May 1960, fighter spectrum to a composite spectrum per the concept contained in MIL-A-8866A (USAF), dated 31 March 1971.

3.3.1.2.1 F-111A Airframe Fatigue Tests. The F-111A airframe fatigue test program was initially scheduled to start April 1965 with the sixth F-111 airframe as the test article. Design changes, weight reduction pro grams, design loads revisions, and a significant

redesign of the F-111B Navy model caused program realignment to assure test of structures representative of fleet aircraft. Also, the late acquisition of the TAC planned usage and the resulting delay in preparation of the fatigue criteria, loads, and spectra contributed to the realignment. Testing started August 1968 with the sixty-seventh airframe as the test article. Program realignment occurred again in 1969 creating parallel test capability of hardware which would minimize down-time impact due to failures and reduce test span time to improve over-all program schedule. The 1969 realignment decision was also influenced by a 1968 wing carry through box failure and the resulting delays necessary for a repair development program plus test article restoration. The fatigue test program was completed July 1974 (Reference Figures C-111.1 and C-111.2, Appendix C-111, Section 6.0).

The test article was a full-scale total airframe (A-4), initially representative of both the F-111A and F-111B with a high percentage of structural components common to both models. However, a major redesign of the F-111B caused reconfiguration of the test article into an airframe representative of the F-111A only. After the 1968 failure of the wing carry through box (WCTB), four WCTB full-scale component fatigue test articles (FW-1-1, 1-2, 1-3, and 1-4) were fabricated to assist design and certification of modifications. During the 1969 program realignment, the full-scale total airframe test article (A-4) was reconfigured into major test components (i.e., wing, horizontal tail, and fuselage/vertical tail). In addition to the total airframe, the pivoting pylon and a total wing with pylons were tested separately.

Initially, the testing objective was to apply 16,000 equivalent flight hours of spectrum loadings derived in accordance with MIL-A-8866 requirements and original design usage (1962/63). This would provide a 4,000 hour safe service life of unrestricted 7.33 g usage. A catastrophic failure of the wing carry through box (A-4 test article) immediately after test start in 1968 and the ensuing development effort for a corrective modification also produced a reassessment of the usage spectrum. The reassessment concluded that the original test spectrum did not accurately simulate expected TAC F-111 usage (relative to available F-100/F-105 Southeast Asia experience).

The test spectrum was revised using TAC crew training syllabus missions and MIL-A-8866, Revision A Mission analysis concepts in association with the F-100/F-105 experience. The spectrum was identified as the Mission Analysis Composite (MAC) spectrum. Because it was not as severe as the original MIL-A-8866 maneuver spectrum "A", application of 24,000 equivalent flight hours of the MAC spectrum loadings were required to demonstrate contract fatigue design requirements compliance. The testing would provide 6,000 hours safe service life of unrestricted 7.33 g usage. This testing objective was achieved on all test articles.

After completion of testing for contract compliance, the fatigue test program was extended by Sacramento ALC funds. Test continuation was done to determine the ultimate fatigue life and identify additional fatigue critical areas. An additional 16,000 equivalent flight hours were applied to the test articles followed by constant amplitude cycling to failure. The 16,000 additional test hours were satisfactorily completed which increased the safe service life to 10,000 hours of unrestricted 7.33 g usage. Table C-111-3, Appendix C-111, Section 6.0. presents the full-scale airframe fatigue test results.

Although a 10,000 hour safe service life was established by the testing, this life for fleet aircraft was contingent upon incorporation of modifications resulting from test failures. The modifications were incorporated into in-production aircraft where possible, but for existing aircraft, a retrofit program for modification was established. The fatigue test results determined the appropriate flight hours for retrofit modification of the individual aircraft.

The two most significant fatigue test failures involved the wing carry through box (WCTB) and the wing pivot fitting (WPF). The WCTB failure occurred August 1968 immediately after the start of testing on the A-4 full-scale test article. The WPF failure occurred April 1970 after completing 12,000 test hours on the A-4 wing major test component.

As these failures along with others were encountered, the resultant modifications were incorporated into the test articles to be qualified by later testing. The modifications which evolved from the fatigue test

program are listed in Table C-111-1, Appendix C-111, Section 6.0.

The fatigue test of the wing/pylon component test article was conducted to evaluate wing hardpoints, fixed pylon structure, and wing structure subjected to extensive external store carriage (anticipated usage in a limited war with conventional weapons). The test article was composed of a left hand outer wing box with pivoting pylon support structure and two fixed pylons attached. Testing began March 1971 and was completed July 1974. A total of 40,000 equivalent flight hours of spectrum loadings was applied to the test article followed by constant amplitude cycling to failure. A safe service life of 10,000 hours commensurate with other testing results was verified.

The pivoting pylon component test was performed to evaluate the pylon structure subjected to extensive external store carriage, and was conducted in association with the wing/pylon component test program. Testing started June 1971 and was completed November 1971 with 40,000 equivalent flight hours of spectrum loadings applied to the test article. A safe service life of 10,000 hours was satisfactorily established.

3.3.1.5 <u>Fracture Control Program.</u> A fracture control program for the F-111 aircraft was not part of the original design approach. The need for fracture mechanics technology surfaced in December 1969 with the loss of an aircraft because of an inflight wing failure due to a flaw in a high-strength steel part of the wing.

At the time the F-111 was conceived and designed, specification design requirements, analyses, and testing for aircraft structure were considered sufficient to provide a safe vehicle. Test articles (and their test programs) were considered representative of fleet aircraft for detecting damage and flaws relative to variation in quality and fleet usage/environment. Coupled with the testing were the quality control programs with nondestructive inspections deemed effective in eliminating non-typical manufacturing flaws.

All of the accepted and proven methods for design and quality control failed to detect the flaw in the wing structure. Furthermore, the F-111 was designed with considerable usage of high strength steel similar to

the flawed part. The need to reaffirm the structural integrity of the F-111 fleet resulted in a fracture control program. The F-111 fracture program also identified the need for similarprograms on all Air Force Weapon Systems (both existing and in development).

The F-111 Fracture Mechanics Program (being a forerunner in the application of the technology to aircraft structure) requires the development of basic fracture mechanics data to allow the program to proceed. Test programs were undertaken along with associated analyses. All of this effort was summarily documented in the three volumes of report FZM-12-13467, "Fracture Mechanics" (Reference Section 7.0, Item 7.3.1.5). The facets of spectrum/environmental testing, operational usage analysis, and risk assessment for fleet assurance were all addressed in this report.

While the fracture mechanics program was evolving after the December 1969 accident, the Air Force and F-111 SPO established a "Recovery Program" to restore confidence in the F-111 structure by inspecting all fleet aircraft. The Recovery Program consisted of disassembly and inspection, ECP/TCTO modifications, and a proof test program, all of which were performed in concert with the fractures mechanics program.

Inspection required a major disassembly to gain access to the critical steel parts being identified by fracture mechanics technology. The inspection examined the parts with regards to quality of fabrication, presence of any cracks, and proper dimensions. Particular attention was given fastener holes (both taper-lok and straight shank). A castastrophic fatigue failure (on the full scale fatigue test article) in August 1968 was determined to have occurred at or adjacent to a taper-lok bolt hole. Although located in a high stress concentration area, the bolt hole had been improperly fabricated.

During the program to develop a fix for the catastrophic fatigue failure, other test failures were incurred in or near bolt holes and were attributed to the quality of fabrication. These events occurred in 1968 and 1969 prior to the December 1969 accident and significantly, involved high strength steel.

The Recovery Program included the incorporation of outstanding ECP/TCTO structural modifications because of the accessibility in the disassembled structure. The modifications were those identified earlier in the F-111 development as required to provide full strength and service life.

The proof test program consisted of a test of the total airframe as an extension of the fracture mechanics program. The test served to screen the structure and particularly critical steel forgings for deficiencies and gross defects, provide a basis for determining safe inspection intervals, and also provide assurance against stress corrosion cracking at interference - fit fastener installations in critical steel parts.

Proof testing occurred after the aircraft had been disassembled, inspected (and refurbished as required), modified viz outstanding ECP/TCTOs, reinspected, and reassembled. The proof test consisted basically of applying a critical design limit load with the airframe at -40 degrees F. The proof test limit loads at 7.33 g's and -2.4 g's were applied at a wing sweep position of LE - 56 degrees. The testing was done in a facility which housed the entire airplane (properly restrained) and was capable of providing a temperature environment of -40 degrees F. Facilities were established at General Dynamics/Fort Worth and at Sacramento ALC, California.

All fleet aircraft completed the proof test program as of 1 August 1970. Two major failures occurred. A left-hand horizontal tail pivot shaft failed at 88 percent of proof test loading, and a lower plate of the wing carry-through structure at 57.5 percent of proof testing loading. The pivot shaft failed because of improper heat treat of a local area of the part. The lower plate failed at an interference bolt hole which had received improper preparation during fabrication.

The proof test concept was continued as a requirement for all newly produced F-111 aircraft. Furthermore, upon establishment of II SIP (second Structural Inspection Program) at Sacramento ALC, all aircraft inspected were required to be proof-tested for a second time (Reference Section 3.6).

3.3.1.6 Stress Corrosion. Stress corrosion cracking (SCC) provided an additional consideration to the fracture mechanics, recovery, and fatigue test programs. Three incidents of SCC during late 1970/early 1971 resulted in a significant program to define their cause(s) and to preclude future occurrences. The stress corrosion cracking program was strongly interfaced with the fracture mechanics program since both programs had a primary interest in the presence of (or the potential for) flaws in the structure.

The incidents of stress corrosion cracking involved two wing pivot fitting (WPF) splice areas (F-111E numbers 16 and 68) and one wing carry-through box (WCTB) (F-111A number 43) in which cracks were discovered in Taper-Lok bolt holes. The cracks in the WPF splices were discovered (1970) by a routine X-ray inspection during the Recovery Program. The crack in the WCTB resulted in a failure (1971) at 58.5% of positive test condition loads during cold proof test of F-111A number 43. The failure during proof test provided the impetus for the SCC program.

The five Taper-Lok holes (one each in the WPF splices and three, in a cluster, in the WCTB) exhibited a total spectrum of failure mechanisms, e.g., fit-up stresses, Taper-Lok stresses, surface contamination (i.e., entrapped coolant/cutting fluids, surface sealant voids, cleaning fluid), and a flaw (s) due to fabrication. (Each hole had a combination of more than one of the mechanisms.) As a result of this evidence of stress corrosion, a series of tests were performed for the purposes of (1) determining the critical flaw size curve at minus 40°F for cracked Taper-Lok holes in D6AC steel plate, (2) determining stress corrosion cracking characteristics of such holes at room temperature prior to cold proof test, and (3) demonstrating that the 115 day aging period of F-111 airplanes prior to proof test was adequate to insure no stress corrosion cracking after proof test. The results of these tests were presented in report FZM-12-13465, "Investigation of Stress Corrosion Cracking and General Corrosion of D6AC Steel at Taper-Lok Fasteners" (Reference Section 7.0, Item 6.3.1.6).

Another factor relative to stress corrosion cracking (and also applicable to the fracture mechanics of Section 3.3.1.5) was general corrosion. Sensitivity of the high strength steel parts to general corrosion (inherent in fleet operational environments) required adherence to maintaining good protection of the parts. Any deterioration of the protection enhanced the chance of a flaw site developing the subsequent appearance of a crack.

Proper fabrication procedures (whether during the initial manufacture or during field repair and/or modifications) were required to prevent built-in flaw sites which operational environments could aggravate into a crack.

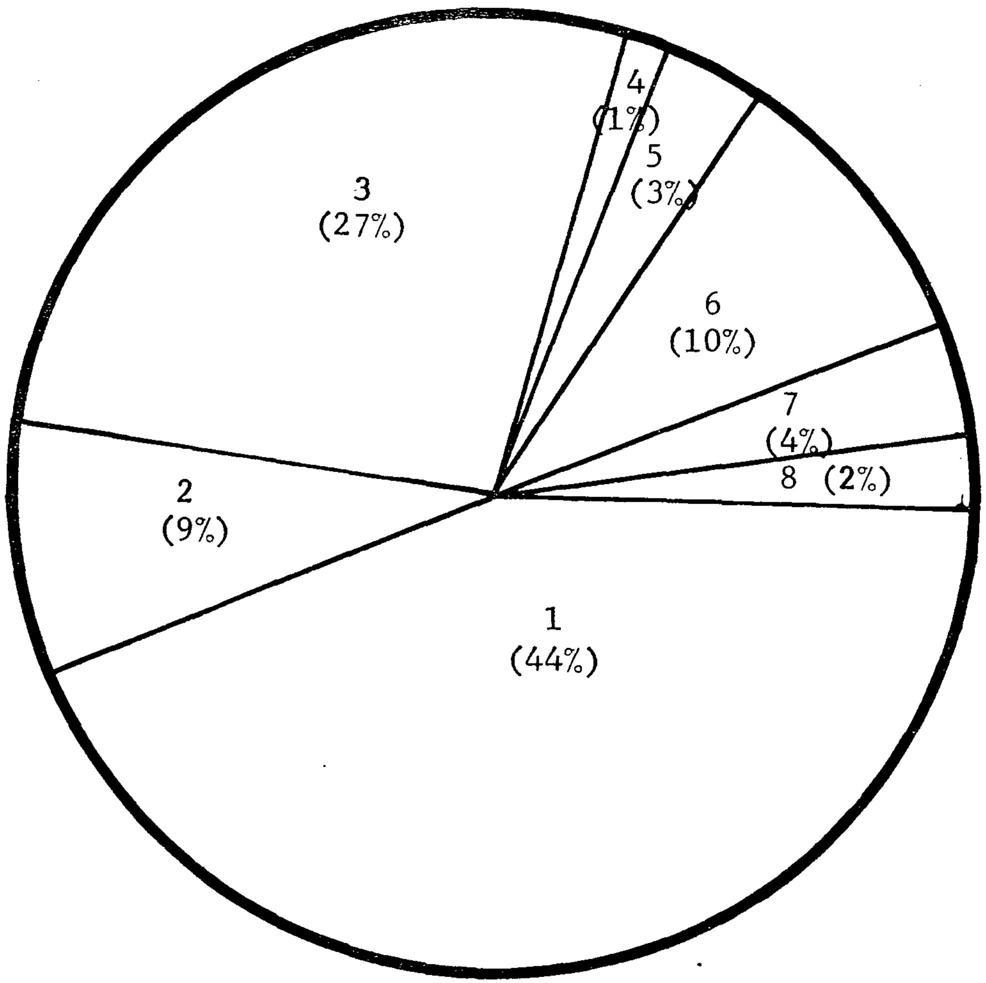
Both surface protection and fabrication aspects (relative to high strength steel parts) have been accented in Tech Order requirements, specifically, T.O. 1F-111-23, Organizational Maintenance - Corrosion Control and T.O. 1F-111-3, Structural Repair Instructions for each F-111 MDS.

Extensive fatigue testing was conducted on the F-111 airframe. This testing included full scale airframe as well as numerous components. Results of these tests are presented in Figures 9 through 14.

Analytical Condition Inspections have been conducted on a scheduled basis for the F-111 aircraft. Results of these inspections for Calendar Years 1975 and 1976 are presented in Table 1 through 4.

In addition to those ACI findings the following F-111 fighter failures have been reported.

- o Horizontal Tail Pivot Shaft Proof Test - Failure caused by under heat treat of D6ac steel pivot shaft.
- o Horizontal Tail Pivot Shaft - Failure caused by hammer peening around an access hole in the D6ac shaft.
- o Failure in the 478 Bulkhead at the speedbrake attachment.



1 120 Incidents of Broken Bolt/Rivet

2 24 Incidents of Receptacle Failure

3 74 Cracking Incidents

4 4 Incidents of Panel Delamination

5 9 Incidents of Shim Failures

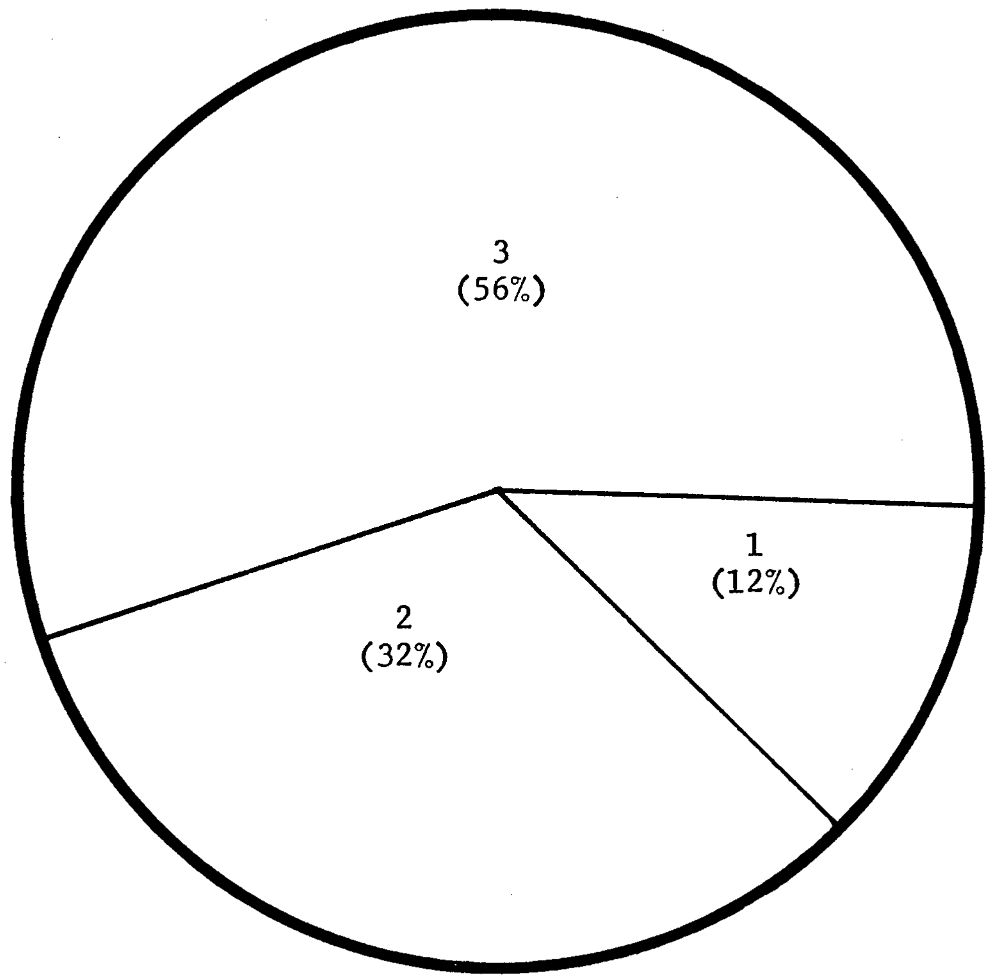
6 26 Incidents of Gouge, Dent, Scratch

7 11 Incidents of Galling, Wear, Hole Non-Roundness

8 7 Other

275 TEST INCIDENTS
(40000 (+) EQUIVALENT FLIGHT HOURS)

FIGURE 9 F-111A (A-4) Fuselage Test Program Summary



1 4 Cracking Incidents Emanating From 4 of 235 TaperLok Holes

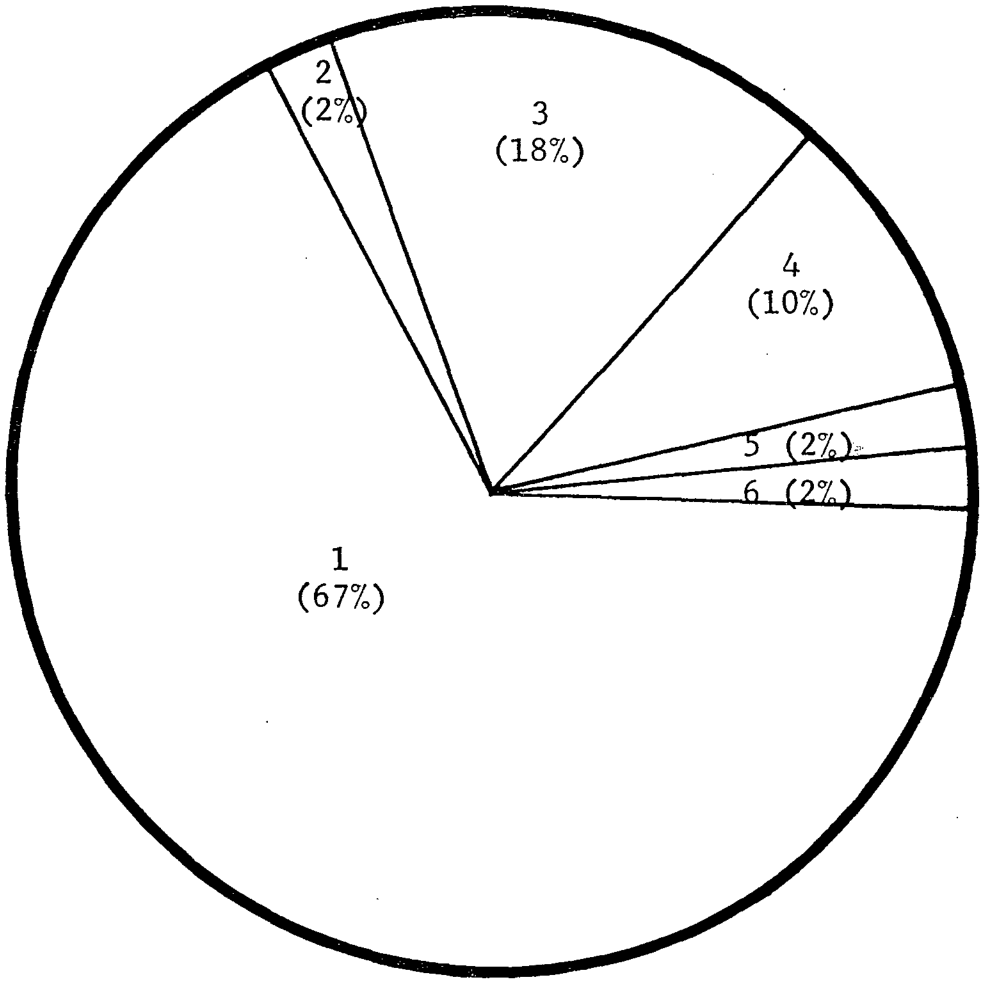
2 10 Cracking Incidents + Catastrophic Failure Emanating From 11 of 24 3/16" Dia. Straight Holes

3 19 Cracking Incidents Emanating from 19 of 1450 Close Tolerance Holes

4 0 Cracking Incidents Emanating From 5 Fuel Flow Cutouts and 1 Fuel Flow Drain Hole

34 TEST FAILURE INCIDENTS
(Block 4 of Planned 40 Block Spectrum)

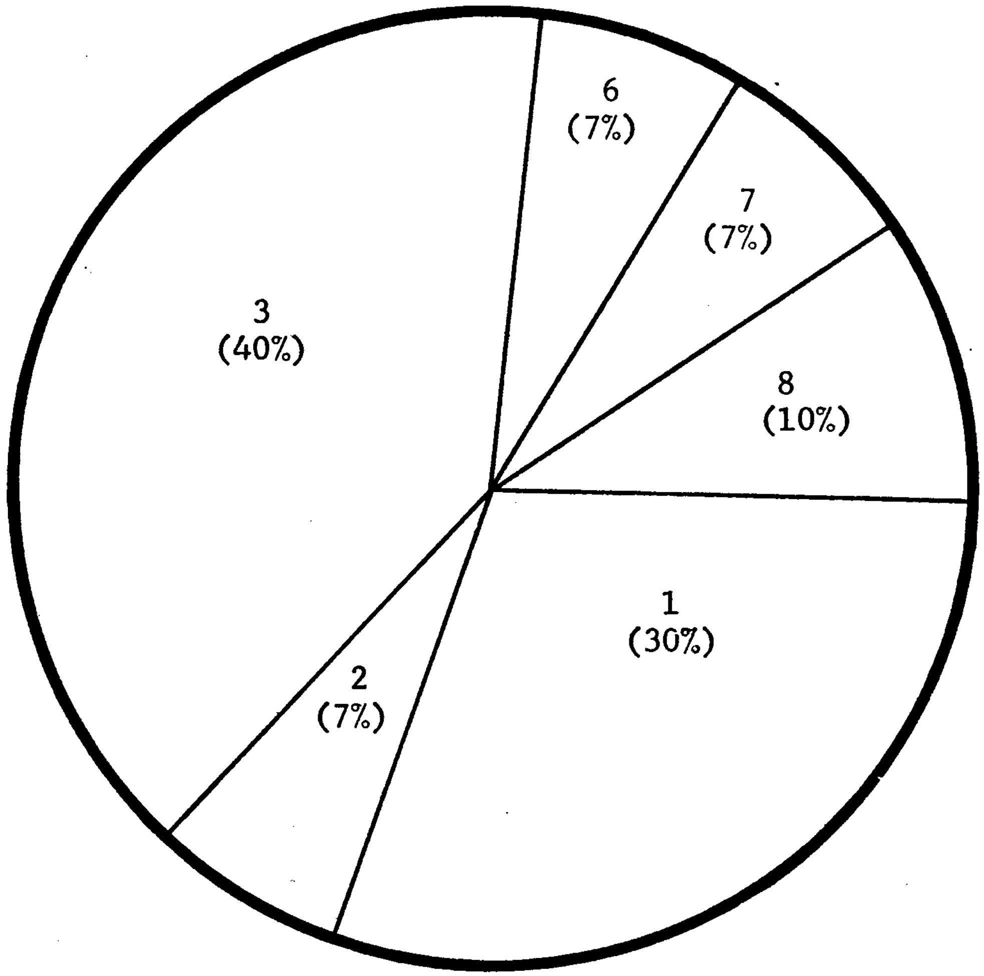
FIGURE 10 F-111A (FW#1) Wing Carry-Through Box Component Test Results



1	34 Cracking Incidents
2	1 Catastrophic Incident Emanating From 1/8" Aft Outboard Flange Corner Radii
3	9 Incidents of Weld Defects
4	5 Incidents of Fastener Failures
5	1 Cracking Incident of Bushing Lip Failure
6	1 Cracking Incident of Shear Bhd Radii

51 TEST FAILURE INCIDENTS
(BLOCK 20 of PLANNED 40 BLOCK SPECTRUM)

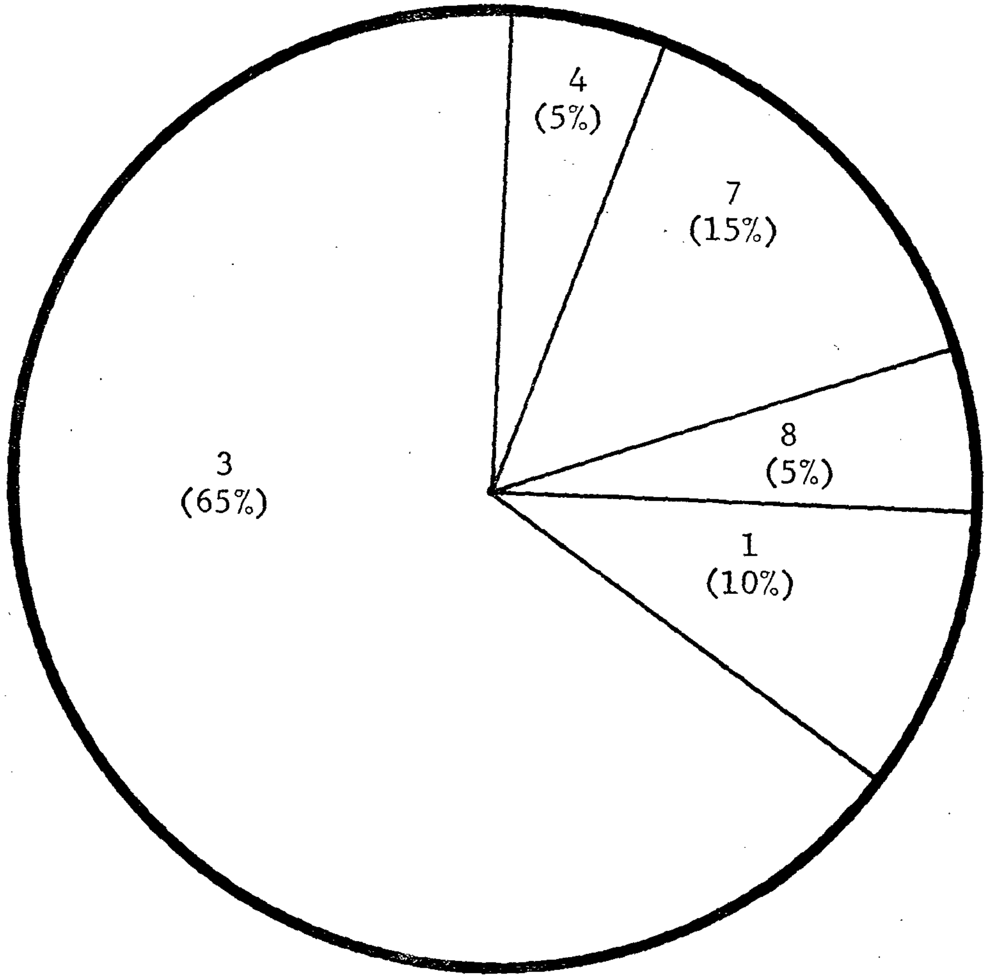
FIGURE 11 F-111A (FW#2) Wing Carry-Through Box Component Test Results



1	9 Incidents of Broken Bolt/Rivet
2	2 Incidents of Receptacle Failure
3	12 Cracking Incidents
4	0 Panel Delamination
5	0 Skim Failure
6	2 Incidents of Galling, Wear, Hole Non-Roundness
7	2 Incidents of Galling, Wear, Hole Non-Roundness
8	3 Other

30 TEST INCIDENTS
(24000 EQUIVALENT FLIGHT HOURS)

FIGURE 12 F-111A (A-4) Vertical Tail Component Test Results



20 TEST INCIDENTS
(40000 (+) Equivalent Flight Hours)

1 2 Incidents of Broken Fasteners

2 0 Receptacle Failure

3 13 Incidents of Cracking Including 1 Catastrophic

4 1 Incident of Panel Delamination

5 0 Shim Failure

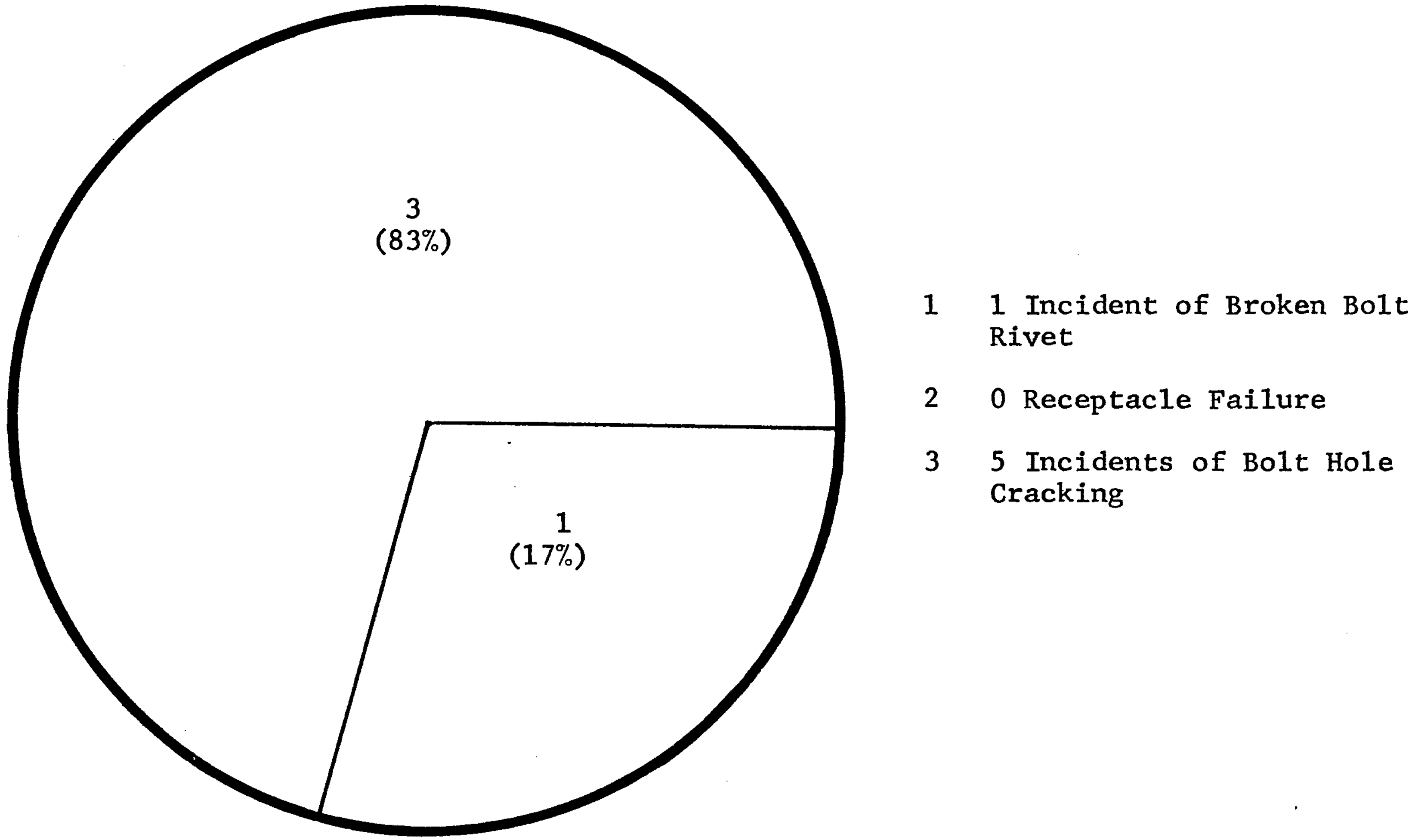
6 0 Gouge, Dents, Scratch

7 3 Incidents of Galling, Wear, Non-Hole Roundness

8 1 Other

NOTE: R/H Wing Failed Catastrophically From 1 Fuel Drain Hole @ 30.5 Blocks of Planned 40 Block Spectra.

FIGURE 13 F-111A (A-4) L/H Wing Test Program Results



6 TEST INCIDENTS

(40000 (+) Equivalent Flight Hours)

FIGURE 14 ECP 10212 Wing Test Results

Table 1 F/FB-111 AIRFRAME STRUCTURAL CORROSION SUMMARY (CY 1975)

AREA DESCRIPTION	NO. AIRCRAFT AFFECTED	GENERAL SEVERITY	PREVIOUS ACI/ SERVICE PROBLEMS	RECOMMENDED ACTION FOR CONTROL/PREVENTION
o CHEEK FRAME @ FS 420.7	1	MINOR	UNKNOWN	ROUTINE MAINTENANCE ATTENTION
o WEAPON BAY - SWAY BRACE	1	MILD	UNKNOWN	ROUTINE MAINTENANCE ATTENTION
o F-1 TANK - 12B2104 BHD @ FS 364	1	MILD	UNKNOWN	MAINT. ATTN DURING DESEAL/RESEAL OPERATION
o CARRY THRU BOX SURFACES (INTERNAL)	2	MILD	UNKNOWN	DLM; II SIP & DESEAL/RESEAL OPERATION
o CARRY THRU BOX SURFACES (EXTERNAL)	2	MILD	UNKNOWN	-6 AND -23 UPDATES REQUIRED
o 12B7356 PYLON ACTUATOR BRACKET	9	MILD/HEAVY	FY 73/74 ACI	-6 UPDATES REQUIRED
o 12B7325 SPEED BRAKE ACTUATOR BRACKET	1	MILD	FY 74 ACI	-6 UPDATES REQUIRED
o NACELLE TIE LINK BEARINGS	7	MILD/HEAVY	FY 74 ACI	DEPOT INSPECT; -6 & -23 UPDATED REQUIRED
o 12B7323 WING SWEEP ACTUATOR FITTING	1	MILD	FY 74 ACI	-6 AND -23 UPDATED REQUIRED
o 496 BHD - NACELLE TIE LINK LUG RADII	3	MILD	FY 73 ACI	-6 AND -23 UPDATES REQUIRED
o 3104/3204 NACELLE PANEL FASTENERS	1	MILD	UNKNOWN	ROUTINE MAINTENANCE ATTENTION
o 12B1831 CENTER FUS. LOWER LONGERON	1	MILD	UNKNOWN	ROUTINE MAINTENANCE ATTENTION
o OVERWING FAIRING HINGE BEARINGS	4	RUST	FY 74 ACI	ROUTINE MAINTENANCE ATTENTION
o 12B1891 LONG - 12B1895/12B1897 FWD SPLICES	8	MILD IN BOLTS & HOLES	FY 74/74 ACI & SERVICE	-23 UPDATE: ENGINEERING STUDY REQUIRED
o AFT FUS UPPER TUNNEL AREA	2	MILD	UNKNOWN	ROUTINE MAINTENANCE REQUIRED
o FS 770 FRAME MEMBERS INSIDE ENG. NACELLE	1	MILD	UNKNOWN	-6 AND -23 UPDATES REQUIRED
o HORIZ. TAIL PIVOT SHAFT (INTERNAL)	5	MILD	SERVICE	ACI SURVEILLANCE ON T.O. ADEQUACY
o HORIZ. TAIL PIVOT SHAFT BUSHINGS	1	MILD	FY 74 ACI	ROUTINE MAINTENANCE ATTENTION
o 12T406 RUDDER TORQUE TUBE	2	MILD/SEVERE	FY 73 ACI & SERVICE	-6 UPDATE REQUIRED
o WING PIVOT FITTING SURFACES o ACTUATOR ARMS o HOLES o UNDER UPPER PLATE BONDED FAIRING	8	MILD	FY 73/74 ACI & SERVICE	-6 UPDATE REQUIRED

TABLE 2 . CY 76 ACI - F-111 AIRFRAME - PRELIMINARY REVIEW

AREA INSPECTED	DEFECT DESCRIPTIONS	REMARKS & ACTION RECOMMENDATION - RELATIVE TO FLEET -	CY 78 ACI ACTION
EQ. BAY DOOR HINGES	MINOR WEAR, SEAL DAMAGE BROKEN LOBES	SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY	DELETED IN CY 77 ACI
EQ. BAY DOOR PANELS	MINOR DENTS, CREASES, DELAM	SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY	CONTINUE
NOSE WHEEL WELL CHEEK PANELS	MINOR SURFACE DAMAGE & HOLE DAMAGE	SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY REF. 183	PANELS INCLUDED WITH GENERAL REQUIREMENT IN CY 77 ACI
AFT NG BULKHEAD	NO DEFECTS	SPECIFIC INSPECTION FOR A SPECIFIC DEFECT	BULKHEAD INCLUDED WITH GENERAL AREA REQUIREMENT IN CY 77 ACI
NOSE WHEEL WELL & CHEEKS-GENERAL	MINOR DOOR HINGE WEAR	SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY	CONTINUE
CHEEK STRUCTURE FS 278-476	NO DEFECTS (CORROSION ON CREW MODULE	N/A DISCONNECTS REPORTED-NOT AIRFRAME STRUCTURE)	CONTINUE
CHEEK FRAMES	NO DEFECTS	N/A - LOOKING FOR STRESS-CORROSION REPAIR RQMNTS.	CONTINUE
CHEEK AREA PANELS	HOLE IN TWO PANELS TEAR IN ANOTHER	SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY	CONTINUE
WEAPON BAY DOORS	DENTS, DELAM SURFACE DAMAGE	SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY	CONTINUE
WEAPON BAY STRUCTURE	CORROSION-GUN & ARMAMENT SUPPORTS	SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY	CONTINUE
FUEL DECK PANELS OVER WEAPON BAY	MINOR DENT	SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY	CONTINUE
INTERNAL STRUCTURE F-1 & LOWER F-2 TANKS	ONE FRAME CRACK & RIVETS FAILED	CRACK - SEE 1E2 RIVETS-SHOULD BE DETECTED & REPAIRED IN DESEAL PGM	CONTINUE
BULKHEADS & FRAMES F-1 & LOWER F-2 TANKS	CRACKS IN FOUR COMPONENTS	DATA USED TO DEVELOP -3 STRESS-CORROSION REPAIRS	CONTINUE
SIDE PANELS F-1 & LOWER F-2 TANKS	WATER PICK DAMAGE	WILL BE DETECTED & REPAIRED IN DESEAL PROGRAM	CONTINUE

TABLE 2 CY 76 ACI - F-111 AIRFRAME - PRELIMINARY REVIEW (Contd.)

AREA INSPECTED	DEFECT DESCRIPTIONS	REMARKS & ACTION RECOMMENDATION - RELATIVE TO FLEET -	CY 78 ACI ACTION
DECK & TOP PANELS F-1 & LOWER F-2 TANKS	WATER PK DAMAGE HOLE IN FLOOR PANEL	WILL BE DETECTED & REPAIRED IN DESEAL PROGRAM	CONTINUE
INTERNAL STRUCTURE UPPER F-2 TANK	NO DEFECTS	N/A	CONTINUE
BULKHEADS & FRAMES UPPER F-2 TANK	NO DEFECTS	N/A-LOOKING FOR STRESS-CORROSION REPAIR RWMNTS.	CONTINUE
EXTERNAL STRUCTURE UPPER F-2 TANK	CRACKED SUPPORT 3411 PANEL	STRESS-CORROSION CRACK. REPAIR DATA TO BE DEVELOPED FOR -3.	CONTINUE
PANELS UPPER F-2 TANK	IFR BOOM DAMAGE	TYPICAL DAMAGE-SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY	CONTINUE
LOWER TRAP TANKS	PAINT PEELING, DAVIS NUT PROBLEMS, DELAM & CORROSION	SHOULD BE DETECTED & REPAIRED INFIELD -6 ACTIVITY	CONTINUE
LOWER FUSELAGE FS 448-480	WATER TANK ← SRAM H_{20} TANK DELAM ←	PDM PROGRAM REFER TO RAF & RBC FOR REVIEW	CONTINUE CONTINUE
SPEED BRAKE HINGE - HOLE MEASUREMENTS	MINOR ELONGATION	QUANTITATIVE DATA PROVIDED. HOLE DIAMETER VARIATIONS INSIGNIFICANT. NO ACTION REQUIRED.	DELETE
INTERNAL STRUCTURE WING CARRY THRU BOX	MINOR SURFACE DAMAGE, PEELING PAINT & CORROSION	SHOULD BE DETECTED & REPAIRED IN DESEAL PROGRAM. ACI WILL EVALUATE NEED FOR LONG RANGE PDM.	CONTINUE
EXTERNAL STRUCTURE WING CARRY THRU BOX	CORROSION -3 AREAS ← LOOSE BRACKET ←	SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY INSIGNIFICANT. WE NEED SPECIFIC -6 & -23 DATA	CONTINUE
WELD JOINTS WING CARRY THRU BOX	NO DEFECTS	MAIN CONCERN IS CORROSION. SHOULD BE NOTED DURING 1H1 & 1H2.	DELETE
PYLON ACTUATOR BRACKETS	ATTACH HOLE CORROSION -14 ACFT	SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY. WE NEED SPECIFIC -6 & -23 DATA.	DELETE
SPEED BRAKE BRACKET	HOLE BURR-INSIGNIFICANT	SLMP CONTROL POINT	CONTINUE
NACELLE TIE LINKS	BEARING CORROSION	INSP. DELETED DURING 76 PROGRAM, SLMP CONTROL POINT. ADD BACK INTO ACI, PLUS 48 MONTH BEARING LUBE. PREFER PDM.	ADD BACK IN

TABLE 2 CY 76 ACI - F-111 AIRFRAME - PRELIMINARY REVIEW (Contd.)

AREA INSPECTED	DEFECT DESCRIPTIONS	REMARKS & ACTION RECOMMENDATION - RELATIVE TO FLEET -	CY 78 ACI ACTION
SHEAR BALLS RECEPTACLES & SHIMS	FABROID WEAR WORN SHIMS	-2 & -6 CHANGES WERE REQUESTED BASED ON CY 75 ACI NEED -2 & -6 REVIEW TO INSURE SHIMS ARE CHECKED AT EVERY WING INSTL.	CONTINUE
WING ACTUATOR SUPPORT FITTING	NO DEFECTS	FASTENER CORROSION IN CY 75 ACI - WE NEED SPECIAL -6 & -23 DATA TO ALLOW FIELD -6 CONTROL.	DELETE
NACELLE DECK PANELS	NO DEFECTS	N/A	CONTINUE
FS 496 FRAME	CORROSION-NACELLE TIE LINK LUGS	SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY. WE NEED SPECIFIC -6 & -23 DATA.	CONTINUE
INLET STRUCTURE	LOOSE/SHEARED FSTNERS FORMER CRACKS SECNDRY DUCT SKIN CRACKS	SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY. NEW PROBLEM-ADD SPECIFIC RQMNT TO 78 PHASE. OLD PROBLEM-MIP SMSRF77-0012-TOM HELBIG	CONTINUE, WITH SEPARATE PHASE REQUIREMENT FOR FORMERS
12B1831 LONGERON	NO DEFECTS	N/A	CONTINUE
EXT. NACELLE PANELS	DELAM (ONE MAJOR ON 3317) HOLES IN PANELS	SHOULD BE DETECTED & REPAIRED IN -6 FIELD ACTIVITY.	CONTINUE
MAIN GEAR BULKHEAD & GEAR ATTACH POINTS	NO DEFECTS	N/A	CONTINUE
UPLOCK BULKHEAD	NO DEFECTS	N/A	CONTINUE
HEAT DAMAGE NEAR HOT AIR LINES	NO DEFECTS	N/A-SHORT TERM EVALUATION FOR EVIDENCE OF HEAT DAMAGE.	DELETE
12B3303 SIDE SHEAR PANELS	NO STRUCTURAL DEFECTS	N/A	CONTINUE
GENERAL WHEEL WELL INSPECTION	UPLOCK ROLLER WEAR BATTERY SCREW CORROSION	SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY.	CONTINUE
OVERWING FAIRINGS	CHAFING & EDGE WEAR BEARING PROBLEMS	REPEAT FROM PREVIOUS ACI's (200 HR -6 RECOMMENDED PREVIOUSLY, -3 REPAIRS EXIST) SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY.	DELETED IN CY 77 ACI
12B1891 LONGERON/FWD SPLICE AREA	MILD CORROSION MOST AIRCRAFT .	-6 & -23 CHANGES INITIATED BY MMSRAM FROM MMET INPUT.	CONTINUE
12B1891 LONGERONS	NO DEFECTS	SLMP CONTROL POINT	CONTINUE

TABLE 2 CY 76 ACI - F-111 AIRFRAME - PRELIMINARY REVIEW (Contd.)

AREA INSPECTED	DEFECT DESCRIPTIONS	REMARKS & ACTION RECOMMENDATION - RELATIVE TO FLEET -	CY 78 ACI ACTION
TUNNEL TRUSSES, BULKHEADS, PANELS	NO STRUCTURAL DEFECTS	N/A	CONTINUE
INTERIOR STRUCTURE A-1 TANK	MINOR DENTS	SHOULD BE DETECTED & REPAIRED IN DESEAL PROGRAM.	CONTINUE
INTERIOR STRUCTURE A-2 TANK	CORROSION-770 BULKHEAD	SHOULD BE DETECTED & REPAIRED IN DESEAL PROGRAM.	CONTINUE
INTERIOR STRUCTURE SADDLE & FINGER TANKS	MINOR DELAM & BRACKET CRACKED	SHOULD BE DETECTED & REPAIRED IN DESEAL PROGRAM.	CONTINUE
EXTERNAL UPPER STRUCTURE & TUNNEL	CANTED LONGERON CHAFING	REPEAT FROM PREVIOUS ACI's (CAUSED BY OVERWING FAIRINGS-1K1) SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY.	CONTINUE WITHOUT SPECIFIC REFERENCE TO CANTED LONGERON
ENGINE NACELLE STRUCTURE	LINER & VAPOR SEAL DAMAGE	SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY.	CONTINUE
725 & 770 FRAMES & ATTACH STRUCTURE	HOLE CORROSION	SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY.	CONTINUE AS CY 77 ACI (INCLUDED WITH NACELLE INSPECTION).
UPPER 770 FRAME & PIVOT FITTING	HOLE CORROSION	SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY.	CONTINUE AS CY 77 ACI (INCLUDED WITH NACELLE INSPECTION).
HOR. STAB PIVOT SHAFTS	INTERNAL SHAFT CORROSION BUSHING GALLING REMOVAL GOUGE	-6 DEPOT INSPECTION-SHOULD BE PDM REQUIREMENT REFER TO FLT CONTROL EXPERTS ISOLATED	DELETED AS SPECIFIC ITEM IN CY 77
ENGINE BAY DOORS & LWR CENTERBODY PANELS	DENT, RIVET PROBLEMS, SURFACE DAMAGE DELAM STRAKE	SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY.	CONTINUE
ENGINE BAY DOOR FASTENER HOLES	NO DEFECTS	ONE-TIME EVALUATION FOR ELONGATION & WEAR. NO ACTION REQUIRED.	DELETED IN CY 77 ACI
AFT CENTERBODY STRUCTURE	FIBERGLASS PANEL HEAT DAMAGE	CORRECTED BY T.O. 1F-111-1117	CONTINUE
HEAT DAMAGE NEAR HOT AIR DUCTS	ONE REPORTED AREA UNDER 3318 PANEL	SHORT TERM EVALUATION	DELETE
V. STAB. & RUDDER EXTERNAL STRUCTURE	RUDDER DELAM	SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY.	CONTINUE

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TABLE 2 CY 76 ACI - F-111 AIRFRAME - PRELIMINARY REVIEW (Contd.)

AREA INSPECTED	DEFECT DESCRIPTIONS	REMARKS & ACTION RECOMMENDATION - RELATIVE TO FLEET -	CY 78 ACI ACTION
V. STAB TIP STRUCTURE	CORROSION AT 4491 PNL COMPASS BRACKET CRACKED	ADD AS PHASED CY 78 ACI REQUIREMENT REFER TO SYSTEMS EXPERTS	CONTINUE
V. STAB ATTACH BOLTS & HOLES	MINOR GALLING & RUST (INFREQUENTLY)	(SLMP CONTROL POINT) ONE TIME SURFACE CONDITION EVALUATION - NO ACTION REQUIRED.	DELETED IN CY 77 ACI. ADD REQMNT TO COVER CONTROL POINT, i.e., NDI
RUDDER TORQUE TUBE	WEAR & ISOLATED GAUGES CORROSION-ACTUATOR ATTACH HOLES	SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY. -WE NEED SPECIFIC -6 DATA.	CONTINUE
HORIZ. STAB. ROOT AREA	BEARING HOUSING CORROSION	-6 DEPOT INSP.-SHOULD BE PDM REQUIREMENT	CONTINUE
WING PIVOT FITTING EXT. & INT. SURFACES	CRACKED BUSHING CHIPPED SHEAR LUG	CORRECTED BY PDM WING JOINT OVERHAUL SHEAR LUG IN -6 NOW FOR CORROSION-SHOULD BE OK	CONTINUE
WING PIVOT FITTING FAIRINGS	DELAM & CORROSION	-PREVIOUS ACI's RECOMMENDED -6 DEPOT INSPECTION -ECP 5022 IN PROCESS FOR RETROFIT NEW DESIGN FAIRINGS.	CONTINUE
WING UPPER & LOWER SKINS	DAMAGE AT T.E. PANEL ATTACH POINT	SHOULD BE DETECTED & REPAIRED IN FIELD -6 ACTIVITY.	CONTINUE
INT. WING STRUCTURE IF SKIN RMVD FOR T.O. 762	NO SKINS REMOVED	ADD FUTURE ACI RWMNT FOR SKIN REMOVAL, PREFERABLY COORDINATED WITH WING DESEAL IF RQD.	DELETE
	PART II - NDI		
	NO SIGNIFICANT FINDINGS, NOR RECOMMENDATIONS, RESULT FROM THE PART II NDI EXCEPT FOR WING SHEAR LUGS.		
NDI-WING SHEAR LUGS	CHROME CRACKS 7 OF 20 INSPECTED	NO INSPECTION ACTION REQUIRED AT PRESENT. o MMSRBA HAD TWO LUGS METALLURGICALLY ANALYZED-CRACKS WERE IN THE CHROME ONLY AND NOT IN THE STEEL. OTHERS ARE IN FOR RECHROMING. o MMSRBA IS EVALUATING FATIGUE TEST SHEAR WGS FOR CHROME CRACKS & EXISTENCE OF CRACK EXTENSION INTO STEEL. o MMIR IS EVALUATING NEW PLATING METHOD.	

TABLE 3 CORROSION SUMMARY - AIRFRAME STRUCTURE - CY 76 - F-111 ACI

AREA DESCRIPTION	AIRCRAFT AFFECTED	GENERAL SEVERITY	PREVIOUS ACI/ OR SERVICE PROBLEMS	RECOMMENDED ACTION FOR CONTROL/PREVENTION
WEAPON BAY-ARMAMENT SUPPORT STRUCTURE	2 A's 1 E	MINOR RUST	CY 75 ACI	FIELD -6 INSPECTION
LOWER TRAP TANK	1 FB	NOT DESCRIBED	UNKNOWN	ROUTINE MAINTENANCE
INSIDE WING CARRY THRU BOX	1 A 1 FB	NOT DESCRIBED	CY 75 ACI	SHORT RANGE: REPAIR DURING DESEAL LONG RANGE: ACI EVALUATION
EXTERNAL WING C.T.B. NACELLE TIE LINK LUGS FWD POST & BOLTS	1 A 2 FB's	NOT DESCRIBED	CY 75 ACI	FIELD -6 INSPECTION
PYLON ACTUATOR BRACKETS	3 A's, 1D, 1E, 9 FB's	NOT DESCRIBED	FY 73/74/75 ACI	FIELD -6 INSPECTION
NACELLE TIE LINK & BEARINGS	1 F	SEVERE	CY 75 ACI	TIE LINK: FIELD -6 INSPECTION BEARINGS: 48 MONTH LUBRICATION
FS 496 FRAME-NACELLE TIE LINK LUGS	2 A's 1 FB	NOT DESCRIBED	SERVICE & FY 73/CY 75 ACI	FIELD -6 INSPECTION
12B1891 LONGERON-FWD SPLICE AREA & BOLTS	4 A's, 1D, 1E, 7 FB's	LIGHT & MILD RUSE-BOLTS & HOLES	SERVICE & PREVIOUS ACI's	FIELD -6 INSPECTION
FS 770 BULKHEAD IN A-2 TANK	1 FB	NOT DESCRIBED	SERVICE: DESEAL RESEAL PROGRAM	SHORT RANGE: REPAIR DURING DESEAL LONG RANGE: ACI EVALUATION
FS 725 BULKHEAD LUG HOLES IN NACELLE	1 A	PITTED HOLE	NEW	FIELD -6 INSPECTION
FS 770 ENGINE FRAME	1 A	MILD RUST IN HOLE	CY 75 ACI	FIELD -6 INSPECTION
FS 770 HOR. TAIL PIVOT SHAFT	2 A's, 1D, 1 FB	NOT DESCRIBED	SERVICE	DEPOT -6 INSPECTION
VERT STAB-FRAME DOUBLER FOR 4491 DOOR	2 FB's	NOT DESCRIBED	NEW	FIELD -6 INSPECTION
VERT STAB-ATTACH BOLT HOLES	1 FB	MILD RUST	NEW	ACI EVALUATION (AS SLMP CONTROL POINT)

TABLE 3 CORROSION SUMMARY - AIRFRAME STRUCTURE - CY 76 - F-111 ACI (Contd)

AREA DESCRIPTION	AIRCRAFT AFFECTED	GENERAL SEVERITY	PREVIOUS ACI/ OR SERVICE PROBLEMS	RECOMMENDED ACTION FOR CONTROL/PREVENTION
RUDDER TORQUE TUBE	1 E 2 FB's	NOT DESCRIBED	SERVICE & PREVIOUS ACI's	FIELD & DEPOT -6 INSPECTION
HOR. STAB. ROOT AREA	1 A, 1 E, 1 F, 2 FB's	NOT DESCRIBED	SERVICE & PREVIOUS ACI's	DEPOT -6 INSPECTION
WING PIVOT FITTING PLATES (UNDER BONDED FAIRING)	1 A	NOT DESCRIBED	SERVICE & PREVIOUS ACI's	LONG RANGE: NEW FAIRINGS PER ECP 5022 INTERIM: DEPOT -6 INSPECTION

TABLE 4 HONEYCOMB STRUCTURE DAMAGE SUMMARY - CY 76 F-111 ACI

STRUCTURE DESCRIPTION	AIRCRAFT AFFECTED	DAMAGE DESCRIPTION	PREVIOUS ACI/SERVICE EXPERIENCE
EQUIPMENT BAY DOORS	4 A's 1 FB	MINOR DENTS, DELAMINATION, PULLED RIVETS	TYPICAL ALL PREVIOUS ACI's
NOSE WHEEL WELL CHEEK PANELS	1 D	2 TORN HOLES	NO PREVIOUS REPORTS ON THIS COMPONENT
WEAPON BAY AREA CHEEK PANELS	1 D 1 E	HOLES IN 2106 & 2104 PANELS EDGE TEAR-LADDER PANEL	TYPICAL DAMAGE, THOUGH NOT PREVIOUSLY REPORTED IN THIS AREA
WEAPON BAY DOORS	3 A's, 1 D, 1 E, 1 F, 3 FB's	MINOR DENTS, DELAM, CORNER DAMAGE & TEARS	TYPICAL SERVICE DAMAGE-PART IS HIGHLY SUSCEPTIBLE TO OPERATIONAL DAMAGE
FUEL DECK PANELS OVER WEAPON BAY	1 A	SHARP DENT	TYPICAL DAMAGE, THOUGH NOT PREVIOUSLY REPORTED IN THIS AREA
UPPER GLOVE PANELS	1 FB	LONG DENT	TYPICAL DAMAGE FROM IN-FLIGHT REFUELING
LOWER TRAP TANKS	1 FB	DELAMINATION	NO PREVIOUS REPORTS IN THIS AREA
EXTERNAL NACELLE PANELS	2 A's, 2 FB's, 1 D	DELAM-3317 & 3319 HOLES IN HONEYCOMB	SIMILAR REPORTS IN PREVIOUS ACI's . 3317 DELAM ON F-111A 67-103 VERY SEVERE.
INTERNAL PANELS A-1 TANK	2 FB's	MINOR DENTS	NO PREVIOUS REPORTS IN THIS AREA
ENGINE BAY DOORS	1 E 6 FB's	MINOR DENTS, PULLED RIVETS, AND STRAKE DAMAGE	TYPICAL ALL PREVIOUS ACI's
RUDDER	1 D	LOWER T.E. CORNER DELAM	TYPICAL SERVICE PROBLEM

- Failure on a lug of the 496 frame where the nacelle tie link attaches. This failure caused by excessive torque inducing a pre-stress in the lug.

- Truss flange cracking on backbone of aircraft - 7079-T6 material.

- Cracking in upper flange on glove bulkhead - 7079-T6 material.

- 12B4811 longeron cracking - 7079-T6 material

- 531 Bulkhead flange cracking - 7079-T6 material

- 1831 Lower steel longeron splice failure due to improper installation.

3.1.8 F-15 Series

The F-15A aircraft is one-place fighter aircraft and the F-15B is a two-place version.

The F-15A/B airplane is a twin-engine, high-wing, supersonic, long-range, all-weather air superiority fighter.

The following list of durability related problems were experienced in the development and early fleet service of the F-15 aircraft. These problems have been corrected for production effectivity:

1. Inlet Duct Skin Panel Cracks
2. Wing Station 206 Upper Skin Cracks
3. Rudder/Actuator Attach Bolt Problems
4. Rudder Leading Edge Spar/Boron Laminating Failures
5. Composite Speed Brake Failure
6. Vertical Stabilizer Cracks
7. Horizontal Stabilizer Cracks
8. Door 110L Cracks
9. Main Landing Gear Strut Door Linkage Support Cracks
10. Inlet Duct Wrinkles
11. Wing Skin Wrinkles

Details of the above durability problems are discussed on the following pages.

- FATIGUE TEST FAILURE

At 4,000 hours of fatigue testing, a crack was detected in the upper wing skin in the inboard area resulting from integral stiffener run-out discontinuities.

The machined inner surface was redesigned to provide increased land thicknesses overlapping the ends of the integral stiffness at the rib attach area in order to reduce the original stress concentrations due to skin span-wise bending. Testing of redesign was successfully accomplished on a small component.

The following current operational problems have been identified:

- INLET DUCT SKIN PANEL CRACKS

These cracks were caused by poor design practice of not providing support in large flat rectangular cross-sections for an acoustic environment. Acoustic levels were also higher than originally had been estimated.

Corrections were made by increasing chem-milled land widths and added cross lands. In some cases standard structural repair techniques were used. Due to magnitude of the problem, a complete analysis and redesign of the subject area was undertaken.

- WING STATION 206 UPPER SKIN CRACKS

Skin cracks were discovered emanating from fastener holes in the upper skin at WS 206 rib on five high time (93-200 hours) aircraft. Cause was an unsupported strip of skin flexing between fasteners with attendant fatigue loading. An external skin patch repair was made on the most severely cracked skins and the skin thickness was increased along with local rib redesign for a production fix.

- RUDDER/ACTUATOR ATTACH BOLT PROBLEMS

Four 1/4 inch bolts which attach each rudder to its actuator drive fitting failed resulting in two separate incidents of rudder failure.

Cause was flush head fasteners yielding due to high loading thus elongating holes and resulting failure.

Solution was to incorporate 5/16 inch protruding bolts in a cold-worked and reamed hole.

- RUDDER LEADING EDGE SPAR/BORON LAMINATE FAILURES

Two occurrences of right rudder outboard boron laminate peel-off.

Insufficient clearances between the lower edges of the rudder leading edge spar, flanges (fiber glass) and fin structure caused this problem. Rudder lateral deflections in service were larger than the clearances provided. As a result, the fin structure contacted the rudder spar flange structure. This caused local failure of the fiber glass flange and exposed the boron laminate skin to the airstream, causing boron plies to be torn away.

Production fix and retrofit was accomplished by trimming back the leading edge of each spar flange in the bottom 12 inches to provide triple the original clearance, wrapping a thin metal sheet around the leading edge of each spar flange and covering the leading edge of the boron/epoxy skin, and installing a fitting the full width of the spar channel cross-section to which both flanges and wrap strips are attached to increase the rigidity of the leading span edge.

- COMPOSITE SPEED BRAKE FAILURES

These structural failures of the composite speed brake occurred on supersonic actuation.

Probable cause was poor adhesion between the lower speed brake skin and aluminum honeycomb core.

Solution was to incorporate a double thickness adhesive between the core and upper and lower skins.

- VERTICAL STABILIZER CRACKS

  Cracking problems have been experienced in three areas of the vertical stabilizer.

  - Chem-milled pocket in vertical stabilizer lower leading edge closure fairing. Production fix was accomplished by increasing gauges of fairing from 0.032 to 0.045.

  - Fillet radii of fairings around light and antenna on right vertical stabilizer. Cracking caused by high vibration environment. Redesign of the upper box assembly to withstand vibration levels has been accomplished.

  - Tip pod support radii cracking where support attaches to tip pod. A production fix incorporating additional straps has been accomplished.

- HORIZONTAL STABILIZER CRACKS

  Cracks were experienced in the inboard lower forward fairings on the horizontal stabilators of five F-15A aircraft undergoing severe service usage.

  These cracks formed along the aft rivet line of the lower triangular aluminum chem-milled panel. Probable cause was panel buffet. Production fix was accomplished by increasing chem-milled bay thickness from .040 to .045 inches and adding additional .071 inch lands.

- DOOR 110L CRACKS

  Cracks have been experienced on the heat exchanger access panel on the aircraft undergoing severe service usage. The cracking of the thin titanium panel resulted from panel flutter. Production fix was to install a stiffener in the chem-milled bay of the panel and add an additional land in the aft bay.

- MAIN LANDING GEAR STRUT DOOR LINKAGE SUPPORT CRACKS

  Cracking has been experienced on an aluminum T-shaped bracket which serves as the door linkage support. This bracket cracked after extended use, leading to further cracking of the adjacent inlet duct floor and MLG trunnion beam. A production and retrofit fix was accomplished by change of material from aluminum to titanium and adding an additional support angle.

- INLET DUCT WRINKLES

  Inlet duct wrinkles have been discovered on several aircraft that had flown high Mach numbers. Cause of these wrinkles was thermal expansion of the inlet duct at high Mach numbers. Critical areas of the inlet ducts have been beefed up for production.

- WING SKIN WRINKLES

  Wrinkles have been experienced in the F-15A/B wing inner torque box upper skin. Cause of these wrinkles was probably from steady state pull up over "g" loads and/or rolling pull-out over "-g" loads. The condition is being monitored and operational units have been briefed on service life implications for such usage.

- F-15A ANALYTICAL CONDITION INSPECTION (ACI) RESULTS

  A list of the defects found during the ACI of F-15A, 73-0085 is presented below, Table 3-5. The accumulated flight hours on this airplane at the time the ACI was conducted was not available.

TABLE 5

DEFECTS FOUND DURING THE ACI OF F-15A, 73-0085

1. Crack approximately 5/8 inch into fastener hole in left wing at W.S. 206.402.
2. Manifold fuel gravity transfer valve installed upside down in fuel tank 3A.
3. Wave washers not installed on any fuel line inside fuel tank 3A and "B" nuts only finger tight at bulkhead fittings.
4. Right engine flame holder cracked in three places.
5. Left engine flame holder cracked in five places.
6. Doubler cracked right side of cockpit under canopy track.
7. Patch on panel 137L.
8. Hinge pin wire frozen in right and left aileron hinge.
9. Hinge pin wire frozen in right and left flap hinge.
10. Five holes double drilled in panel 64L.
11. Three holes double drilled in panel 132R.
12. Three holes double drilled in panel 132L.
13. Eighteen fasteners missing from engine access door 95L.
14. Four fasteners missing on engine access door 113R.
15. Five fasteners missing from engine access door 95R.
16. Nine fasteners missing from engine access door 117L.
17. Fifteen fasteners missing from engine access door 115L.
18. Right inlet first ramp top access door chafing screw heads in panels 19R and 20R.
19. Left inlet first ramp top access door chafing screw heads in panels 19R and 20L.
20. Paint chipped and peeling throughout aircraft.

3.1.9 F-16 Series

The F-16A is a single-engine, single-seat, multirole tactical fighter with full air-to-air and air-to-ground capabilities.

The F-16B is a two-place fighter/trainer version of the basic F-16A aircraft. It has full combat as well as training capabilities.

The service life criteria for the F-16 are based on the requirements of MIL-A-008866A, MIL-A-83444 and MIL-STD-1530. The requirements of MIL-A-83444 are applicable to the airframe only.

The F-16 design usage is established as 8000 flight hours, 5754 sorties, and 6555 landings (801 of which are touch and go landings), with a 15-year service life.

The materials and processes specified for the F-16 air combat fighter were selected to provide compatibility of design, manufacturing, and assembly at minimum cost while maintaining design integrity and reliability. Special attention was given to overcoming material problems encountered in the past. New and improved materials/processes which avoid durability problems such as stress corrosion and brittle fracture, coupled with a design philosophy which recognizes such problem areas, have resulted in a design that will provide longer life with improved performance at lower maintenance cost. Figure 15 presents the F-16 structural arrangement and material selections.

A full scale durability test program was conducted for the F-16 airframe. The airframe was tested to two service lives (8000 hours each) followed by a teardown inspection. Results of the two 8000 hour tests are shown in Figures 16 through 20. Teardown inspection results are shown in Appendix B.

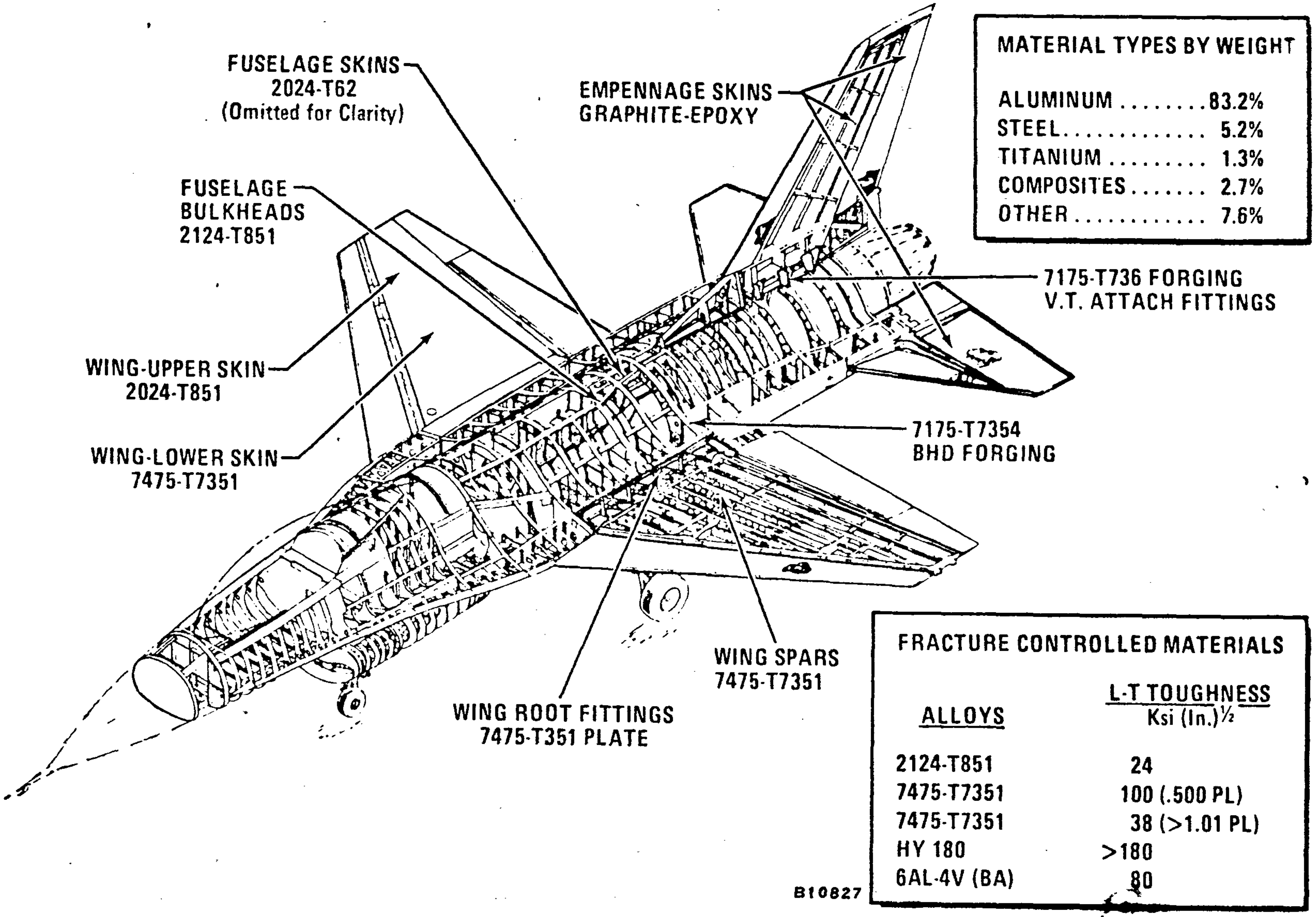
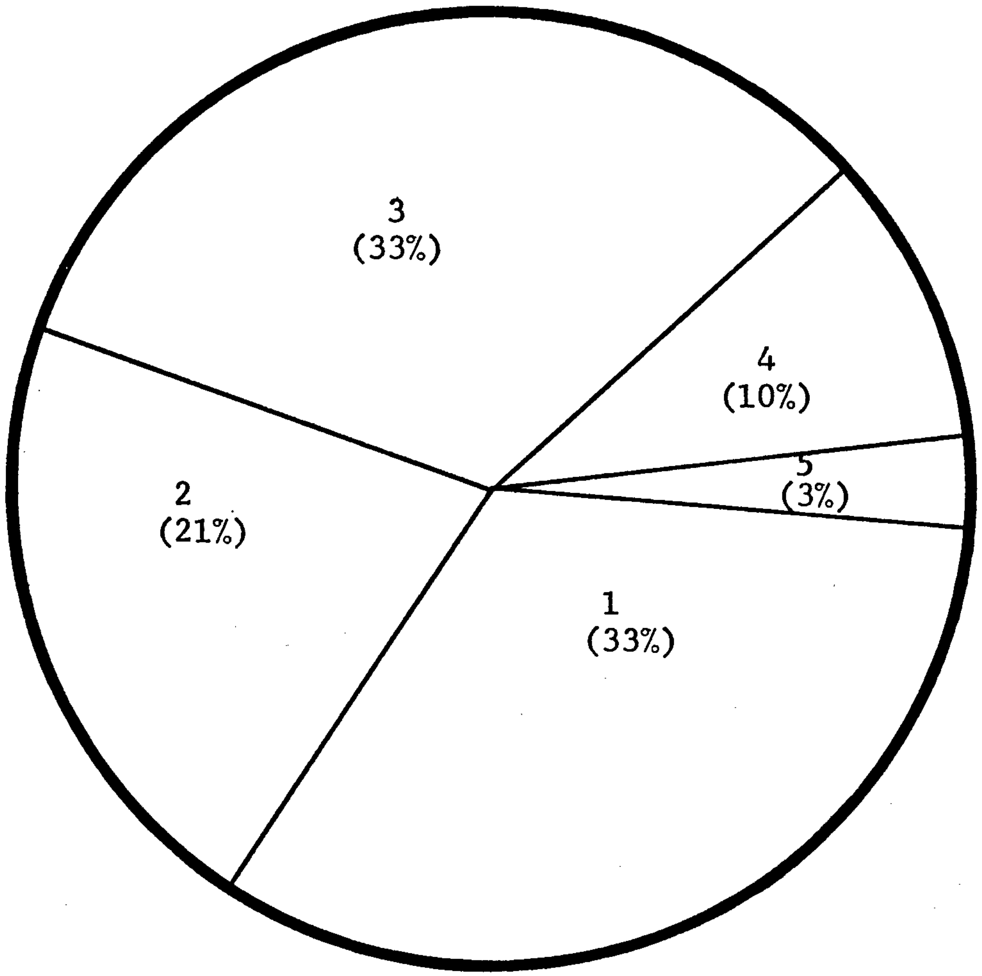


FIGURE 15 F-16 Structural Arrangement and Materials.



1 20 Cracking Incidents Emanating from Radii

2 13 Cracking Incidents Emanating from Fastener Holes

3 20 Cracking Incidents Emanating From Bhd Web Access Holes

4 6 Cracking Incidents Emanating From Cutouts

5 2 Other

<u>NOTE</u>: All Parts Experiencing Cracking Were Redesigned For Production Configuration

61 TEST FAILURE INCIDENTS
(1st 8000 HOUR SERVICE LIFE)

FIGURE 16 F-16 Durability Airframe Test Results For 1st 8000 Hour Life

F-16 DURABILITY TEST RESULTS AT 8000 HOURS INCLUDING 8000 HOUR UPDATE

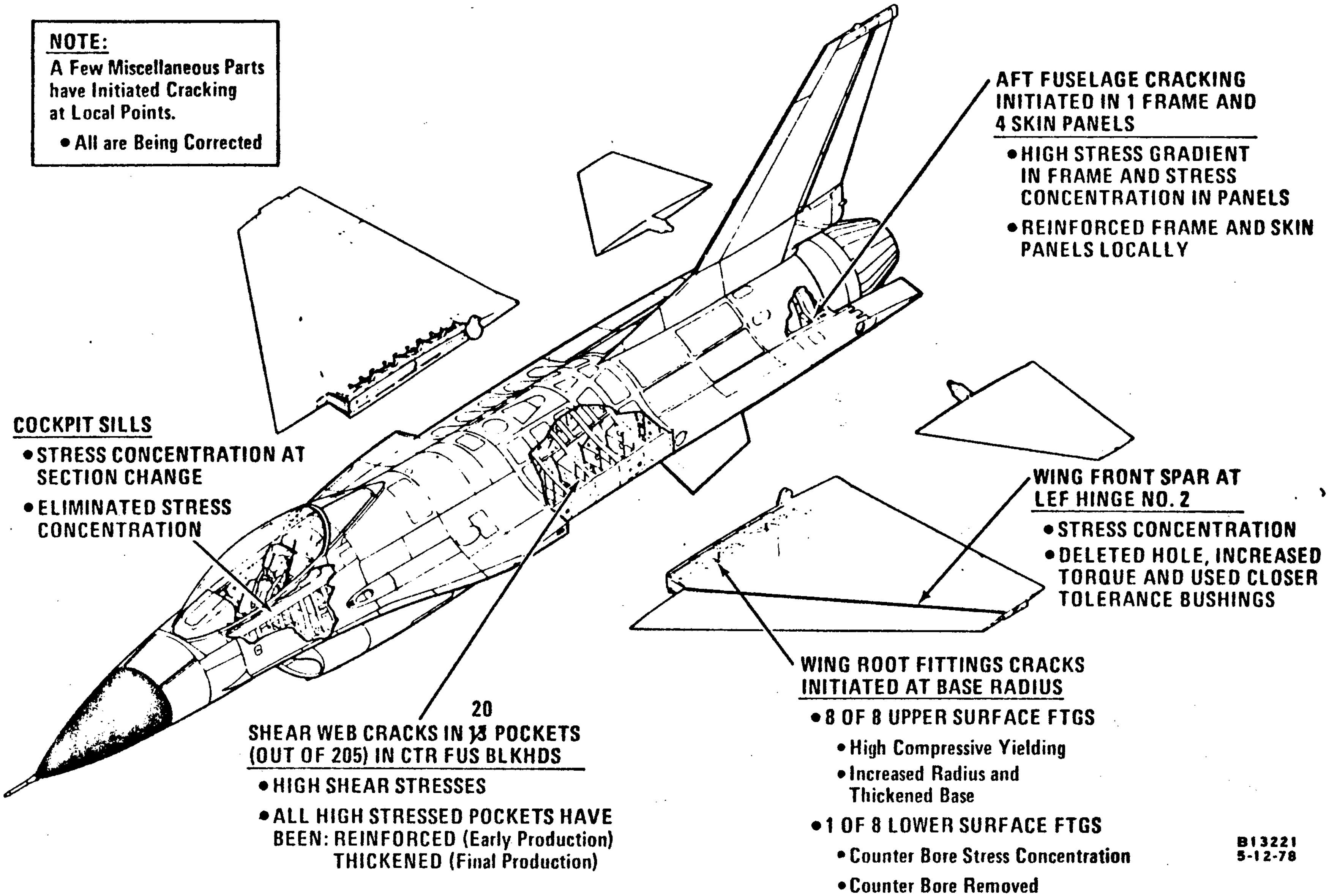


FIGURE 17 F-16 Durability Test Results at 8000 Hours Including 8000 Hour Update

MODIFICATION TO TEST ARTICLE – 8000 HOURS

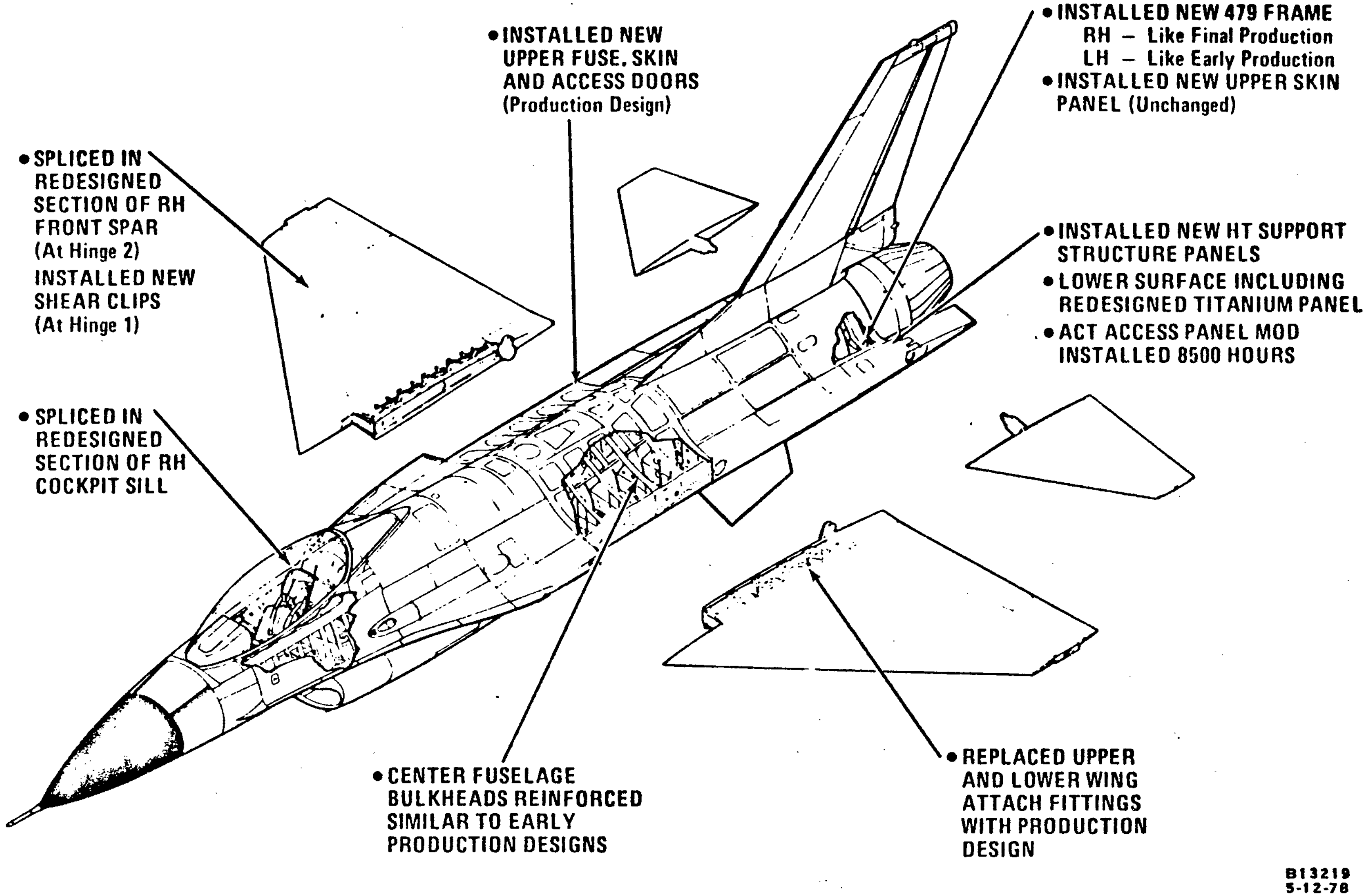
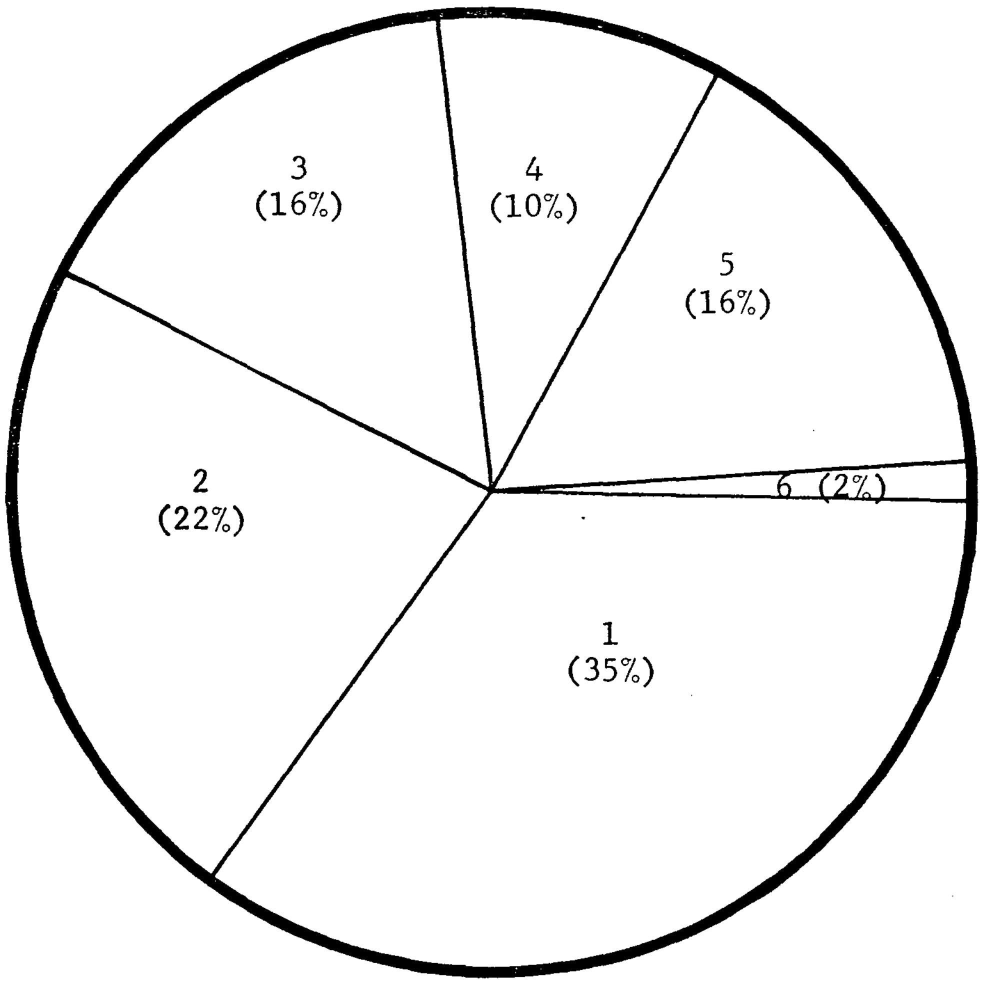


FIGURE 18 ModificationsTo Test Article - 8000 Hours



63 TEST FAILURE INCIDENTS
(2ND 8000 HOUR SERVICE LIFE)

1 22 Incidents of Bolt Failures

2 14 Cracking Incidents Emanating From Bhd Web Access Holes

3 10 Cracking Incidents Emanating From Fastener Holes

4 6 Cracking Incidents Emanating From Cutouts

5 10 Cracking Incidents Emanating From Radii

6 1 Crack Incident Emanating From Large Hole

NOTE: All Parts Experiencing Cracking Were Redesigned for Production Configuration.

FIGURE 19 F-16 Durability Airframe Test Results For 2nd 8000 Hour Life

F-16 DURABILITY TEST RESULTS DURING 2ND 8000 HOURS

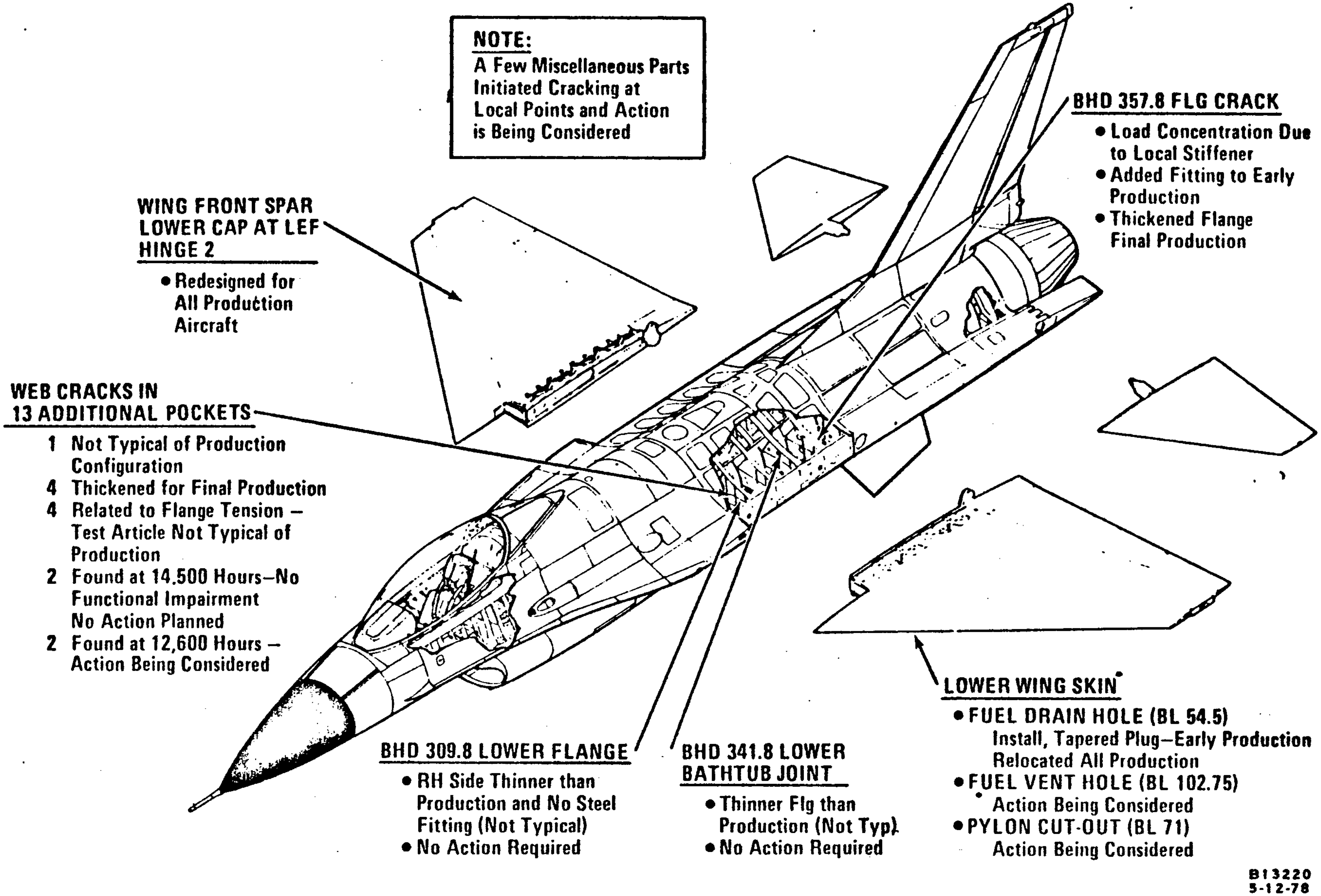


FIGURE 20 F-16 Durability Test Results During 2nd 8000-Hour Life (Schematic)

3.2 TRAINER AIRCRAFT

3.2.1 T-37 Series

The USAF T-37B is a low wing, two place, side-by-side arrangement jet aircraft.

The mission of this airplane is to serve as a primary trainer for elementary contact and instrument maneuvers to qualify the students for continued extensive pilot training.

The T-37C conforms with the mission and configuration of the T-37B except for added fuel tanks and armament.

Among the more significant durability problems encountered on the T-37 airplane was:

- o WING FRONT SPAR
  - Aircraft accident caused by fatigue failure of wing front spar
- o WING
  - Catastrophic wing failure caused by fatigue
- o ELEVATOR
  - Inflight failure caused by fatigue
- o CANOPY RAIL
  - Fatigue test failure of the canopy rail attachment fitting generated a modification for this area.

Details on the above durability problems have been documented; however, the reports were not readily available for this report. The reference 4 document identifies those fatigue reports documenting the above problems.

3.2.2 T-38 Aircraft

The T-38 is a two-place, twin-turbojet supersonic trainer. This airplane is the mainstay of the Air Training Command Undergraduate Pilot Training. These fleets are used in various roles, such as the Lead-in-Fighter Training, the Thunderbirds, the NASA and USN test pilot training and as chase aircraft.

Durability problems common to the T-38 trainer aircraft are listed below.

- o WING
  - Fatigue test failure of lower wing skin originated in fuel drain hole. Subsequent redesign relocated the drain hole.
  - Fatigue test failure of 66% wing spar also necessitated redesign.
- o GENERAL OBSERVATIONS
  - Poor factory workmanship
  - Poor field workmanship
  - Replacement wing - poor quality control
  - Short edge distance on spar caps
  - Stress corrosion in 7075-T6 and 7079 materials
  - Canopy rail strikes
  - Wing-fuselage mating mismatch
  - Frame cracking around access panel due to frame stiffness
  - Secondary structural failures of intake ducts
  - Bonding, debonding and chem milling problems
  - Nutplate rotation
- o TEARDOWN INSPECTION OF TWO HIGH TIME AIRCRAFT
  - Fatigue cracking in aileron pulley bracket.
  - Stress corrosion in aluminum die forgings of root rib, 66% spar, 44% spar and 21% center section spar

- ANALYTICAL CONDITION INSPECTION (ACI) RESULTS

Analytical condition inspections were performed on 81 aircraft. Six significant structural problems were uncovered.

1. Wing spar and rib stress corrosion cracks.
2. Wing lower skin corrosion.
3. Fuel cell floor cracks.
4. Honeycomb failures in the trailing edge of the flap, wing tip, wing leading edge and horizontal stabilizer.
5. Worn and loose wing attach bolts.
6. Stress corrosion cracks at the F.S. 287.223 bulkhead at the intersection with the intermediate longeron.

- SPECIAL INSPECTIONS

Problem areas for which special inspections have been conducted are:

1. Stress corrosion cracks in the 66T wing spars.
2. Landing gear uplock rib fatigue cracks.
3. Stress corrosion cracks in the wing rib.
4. Canopy panel attachment problems.
5. Surface condition in the fuselage intersection cut-out in the wing lower skin.

- WING DAMAGE TOLERANCE ANALYSIS

The wing damage tolerance analysis identified the following problem areas:

- Fatigue critical fastener and drain holes.
- Wing root radius at the 44% spar on the lower wing skin.

o T-38 TEARDOWN INSPECTION RESULTS

Due to the susceptibility of the T-38 lower wing skin to fatigue cracking and the advance age of the fleet, a teardown inspection was conducted on three high time Air Training Command (ATC) wings.

The significant findings from this inspection are presented in Appendix B to this report. These results were extracted directly from the Reference 5 final report.

3.2.3 T-39 Series

The T-39 Sabreliner is a low wing, twin-jet monoplane with an axial-flow pod-mounted engine on each side of the aft fuselage. There are two versions of the Sabreliner - the commercial Sabreliner, and the military version, designated T-39 aircraft.

T-39 aircraft are used by the Air Force for training and transportation of passengers and cargo.

Durability problems that have been associated with the T-39 aircraft are:

o CURRENT PROBLEMS

1. Corrosion is the most significant problem - frames and longerons.

2. In flight service failure of the main entrance upper door stop.

3. In service stress corrosion failures of fuselage stiffeners at FS 333.6.

4. In service cracking of fuselage frames from FS 465 to 505 cause by vibrations of electrical bundles and vent lines attached to the frames.

o ANALYTICAL CONDITION INSPECTION

An in-depth ACI was performed on three T-39A aircraft at Sacramento ALC. This ACI showed that because of inadequate field level maintenance, the three aircraft were in poor condition. Some of the structural failures found in the ACI could cause loss of the aircraft.

The three aircraft selected for the ACI were representative of the T-39 fleet from both environmental usage and accumulated airframe hours. The airframe hours range from a minimum of 8470 to a high time of 9830 hours.

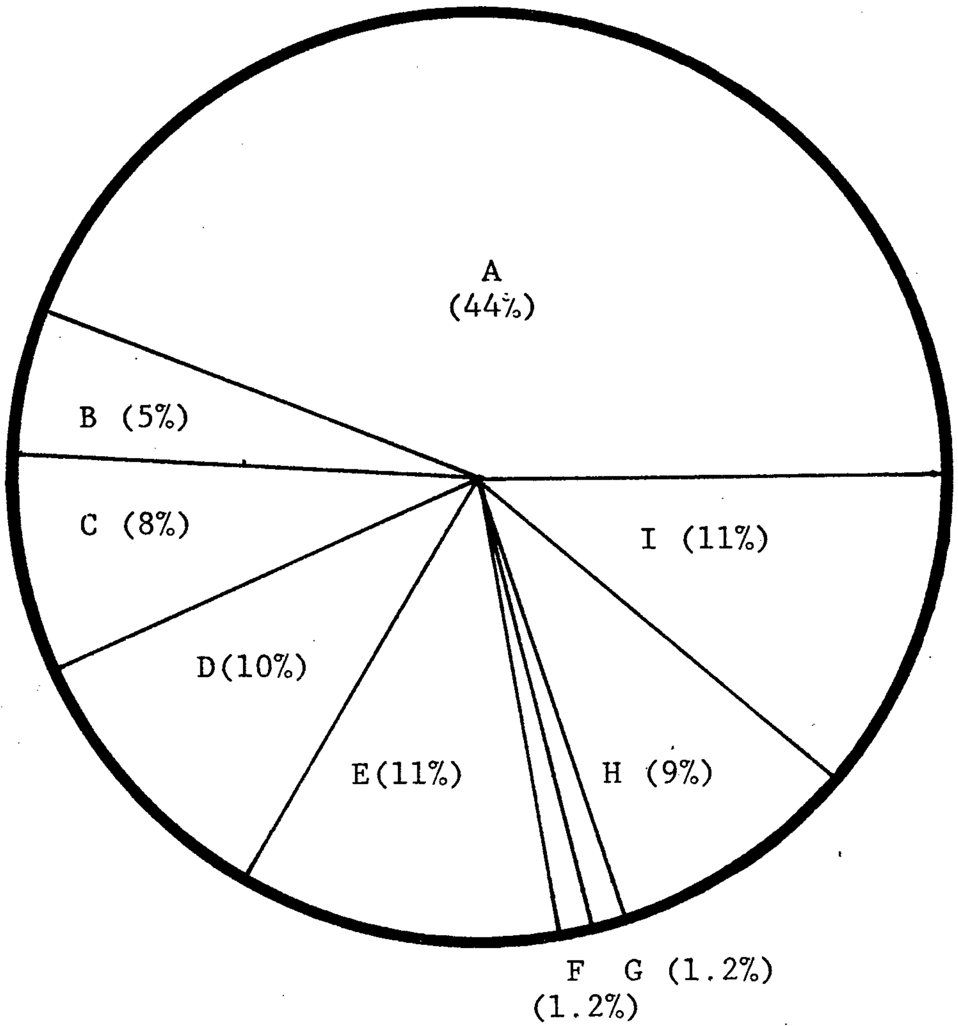
A composite summary of these findings is presented in Table 6.

TABLE 6 T-39A ANALYTICAL CONDITION INSPECTION RESULTS

DESCRIPTION OF DAMAGE	CUMULATIVE OCCURRENCES
o Cracking	142
o Stress Corrosion Cracking	16
o Corrosion	25
o Fastener - missing/loose	31
o Dents/Nicks/Scratches	36
o Honeycomb - Damage & Delamination	4
o Enlarged or Worn Fastener Holes	4
o Wear (Excessive) - Chafing - Maintenance Induced	29
o Maintenance Related - Improper Maintenance - Improper Mods - Bending etc.	37

Figure 21 ranks the results in percentage by number of occurrences for the composite findings.

Figures 22 and 23 show the problem areas for two different aircraft.



324 Incidents

A	142 Incidents of Cracking
B	16 Incidents of Stress Corrosion Cracking
C	25 Incidents of Corrosion
D	31 Incidents of Loose or Pulled Out Rivet/Fasteners
E	36 Incidents of Dents/Hicks/ Scratches
F	4 Incidents of Honeycomb Damage/Delamination
G	4 Incidents of Fastener Hole Enlargement or Wear
H	29 Incidents of Wear, Chaffing Chaffing or Maintenance Induced
I	37 Maintenance Related Incidents; Improper Maintenance, Improper Modification or Damage Induced During Maintenance

FIGURE 21 T-39A Analytical Condition Inspection
(3 Aircraft Surveyed)

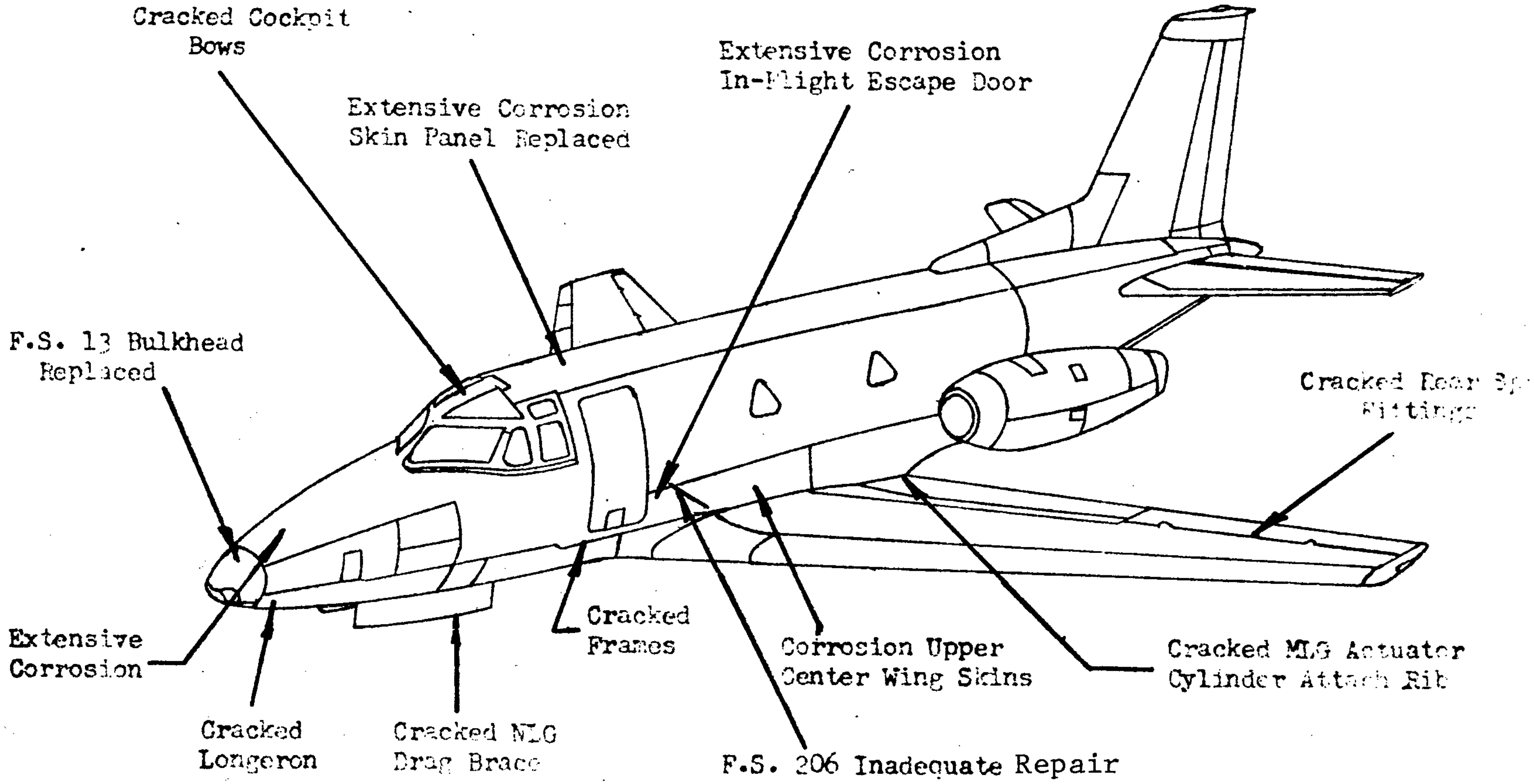


FIGURE 22 ACI FAILURE AREAS T-39A S/N 60-3490 PACAF/TAN SON NHUT

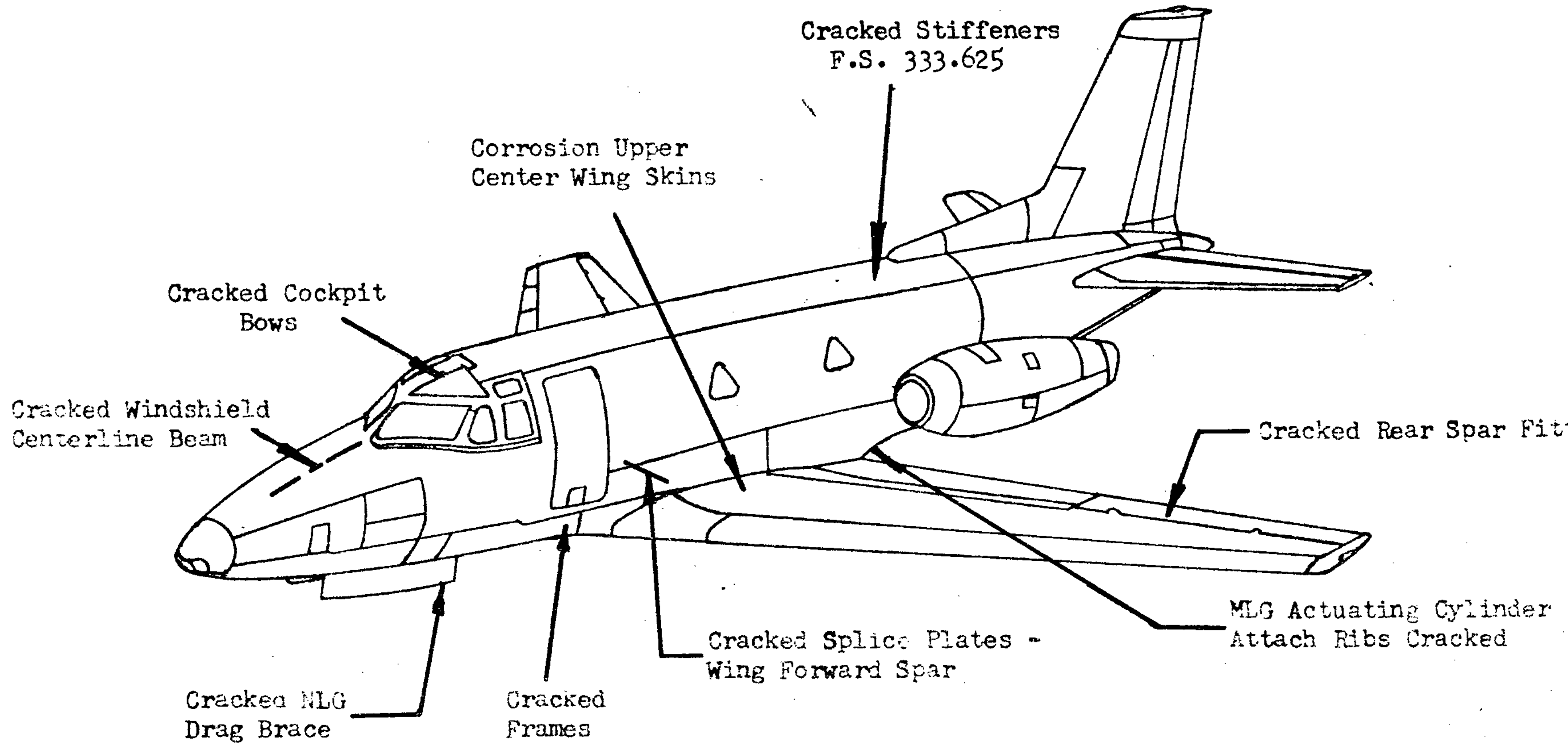


FIGURE 23 ACI Failure Areas T-39A S/N 60-3508
HQC/Andrews AFB

- T-39 WING FATIGUE TEST TEARDOWN INSPECTION

A teardown inspection was conducted on a T-39D wing that had accumulated 4250 service hours. Subsequently, a flight-by-flight spectrum was applied and at 32,000 test hours a failure initiated at a bolt hole on the forward edge of the lower skin. Reinforcing doublers were added and the test was continued to 75,000 test plus service hours.

Subsequent to the completion of the test, the wing was torn down and non-destructively inspected for cracks in fastener holes. All holes in the upper and lower wing skins, splice plates, repair doublers and spar flanges were inspected using the automatic eddy current scanning system.

Figure 24 presents a histogram of the crack sizes recorded from the above L/R wing teardown inspection as documented in Reference 3, NA-77-599-1.

- T-39 FUSELAGE AND VERTICAL TAIL TEST TEARDOWN INSPECTION

A teardown inspection was conducted on an NA-265-40 Sabreliner fuselage that had been subjected to a block type spectrum of fuselage bending loads for an equivalent 90,000 flight hours including a total of 43,000 pressurization cycles. The teardown was concentrated in the pressurized cabin area but also included the vertical stabilizer rear beam and engine pylon attach fittings. Also included in the teardown were the major longerons, wing attach frames and fittings, and the windshield support beam. Details of this teardown are given in Reference 3, NA-77-599-1.

A histogram of the fuselage crack sizes is shown in Figure 25 for the frames, skins and longerons. Figure 26 presents the vertical stabilizer crack size distribution.

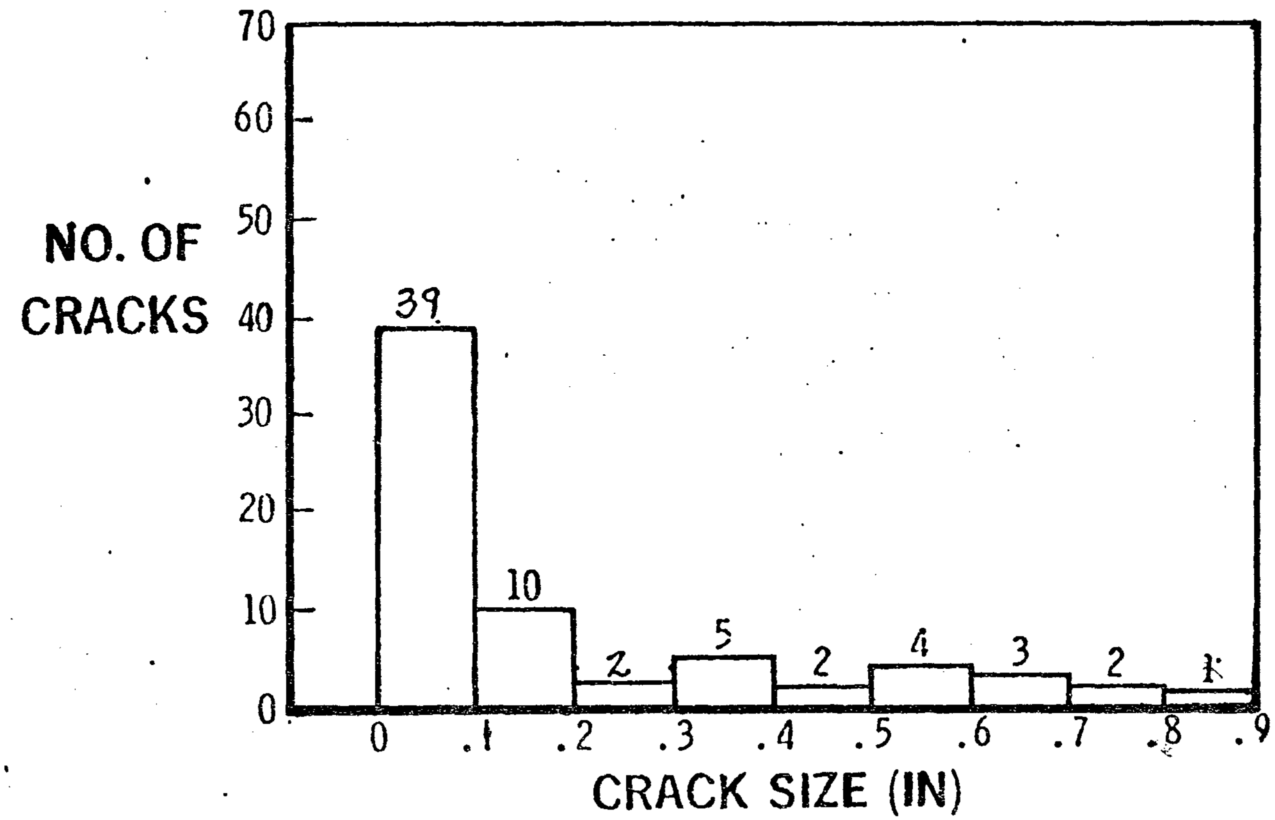


FIGURE 24 Histogram of Wing Crack Sizes (Source: T-39D S.L.E.P. Wing Teardown Report NA-77-595-1)

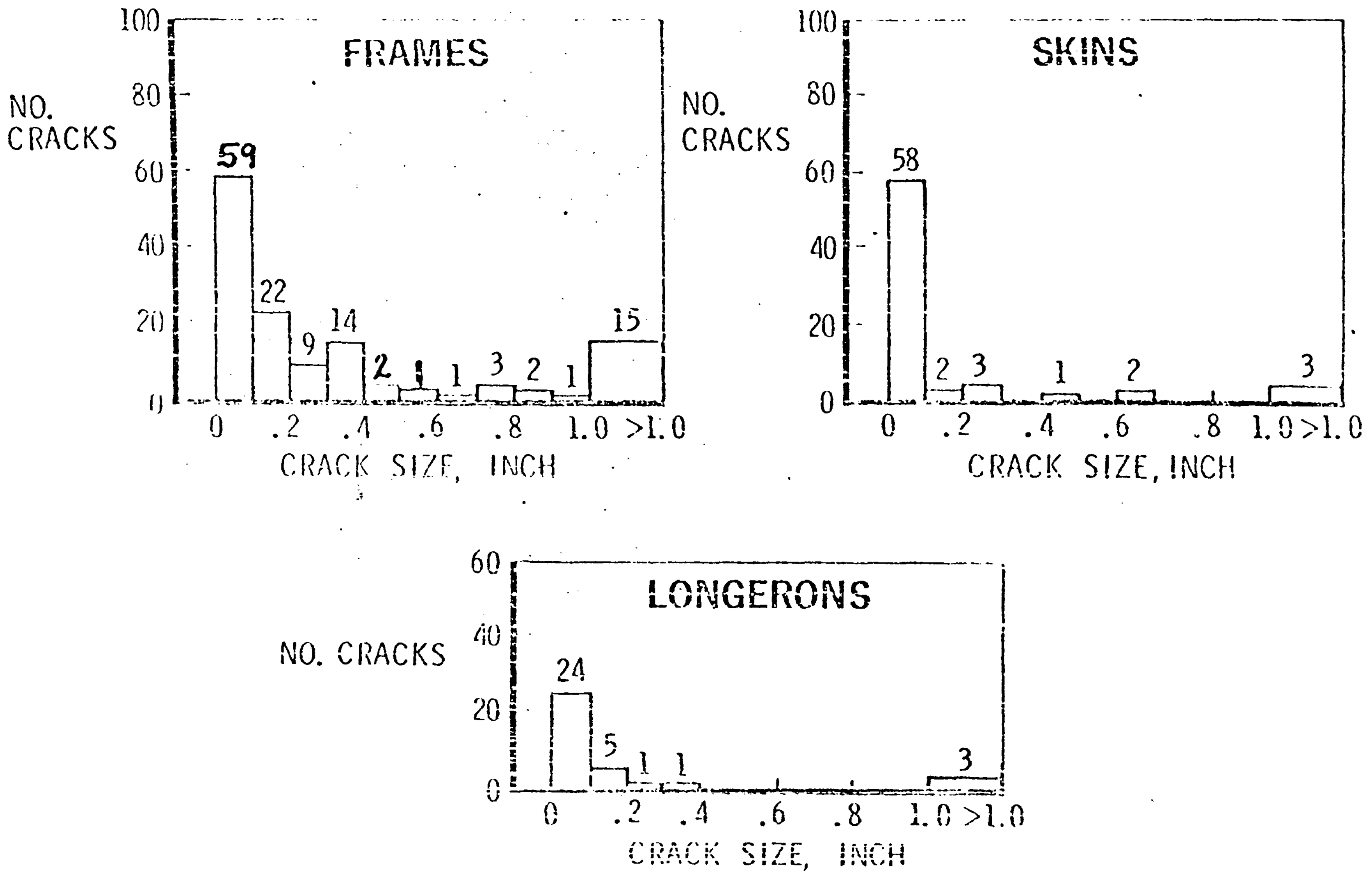


FIGURE 25 Histogram of Fuselage Crack Sizes
(Source NA-265-40 Fuselage
Teardown Report NA-77-595-2)

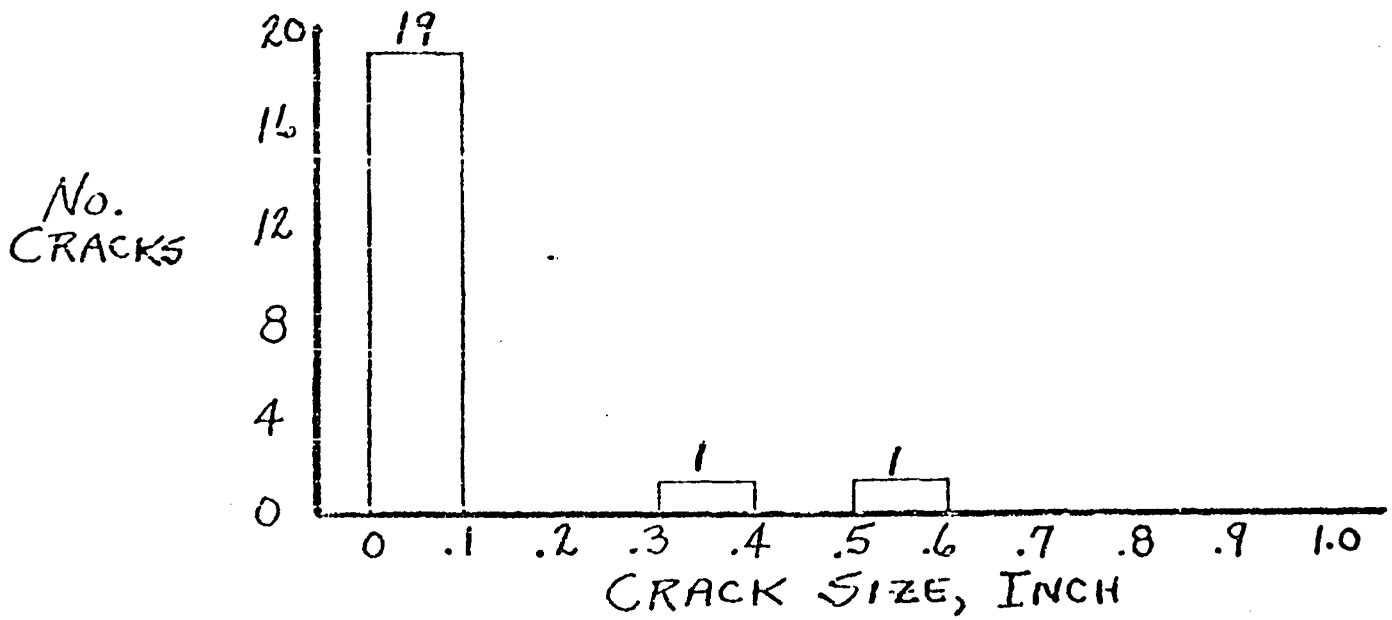


FIGURE 26 Histogram of Vertical Stabilizer Crack Sizes (Source: NA 265-40 Fuselage Teardown Report NA-77-595-2)

o T-39 SERVICE CRACKING HISTORY

A formal T-39A force inspection program was undertaken as part of the T-39A Remanufacturing Program during 1971-1973. Twenty-two specific locations were inspected on 130 aircraft when they reached the 10,000 flight hour level.

A gross quantity of 166 cracks in the wing structure and 1368 cracks in the fuselage were found on the 130 aircraft. No single location was found to have a high incidence of cracking, the crack total being well distributed over the entire area inspected.

A comparison of the wing service failures found during this inspection is made with the cracks identified during test and the subsequent teardown inspection. This comparison is shown in Figures 27 and 28 .

The service cracking history logs from the Remanufacturing inspection program were of limited value from a qualitative standpoint since the specifics of crack lengths were not recorded.

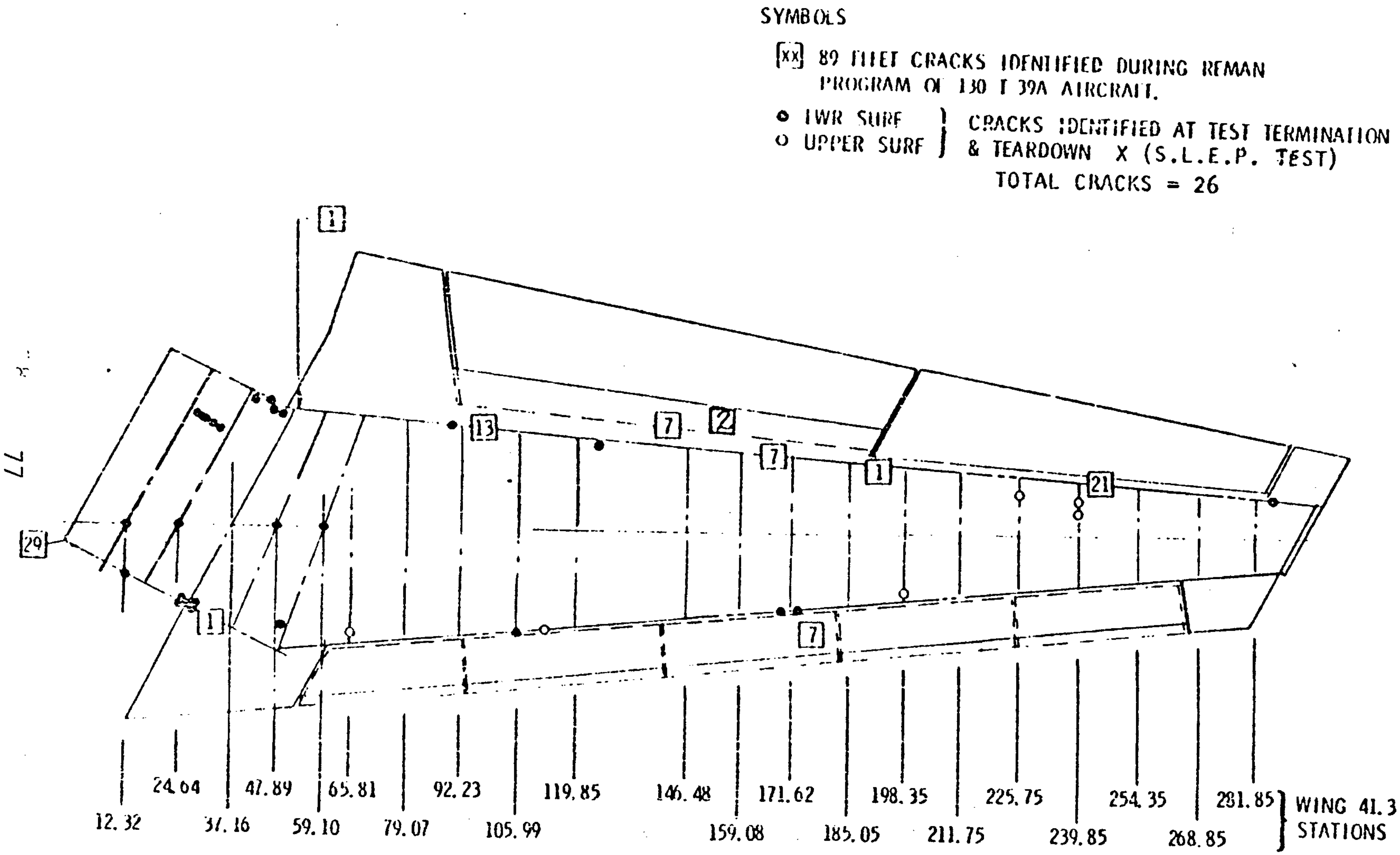


FIGURE 27 T-39A Left Wing Crack Distribution-
Service & Test

SYMBOLS

[xx] 77 FLEET CRACKS IDENTIFIED DURING REMAN PROGRAM OF 130 T-39A AIRCRAFT.

● LWR SURF. } CRACKS IDENTIFIED AT TEST TERMINATION (S.L.E.P. TEST)
○ UPPER SURF. } & TEARDOWN   TOTAL CRACKS = 42

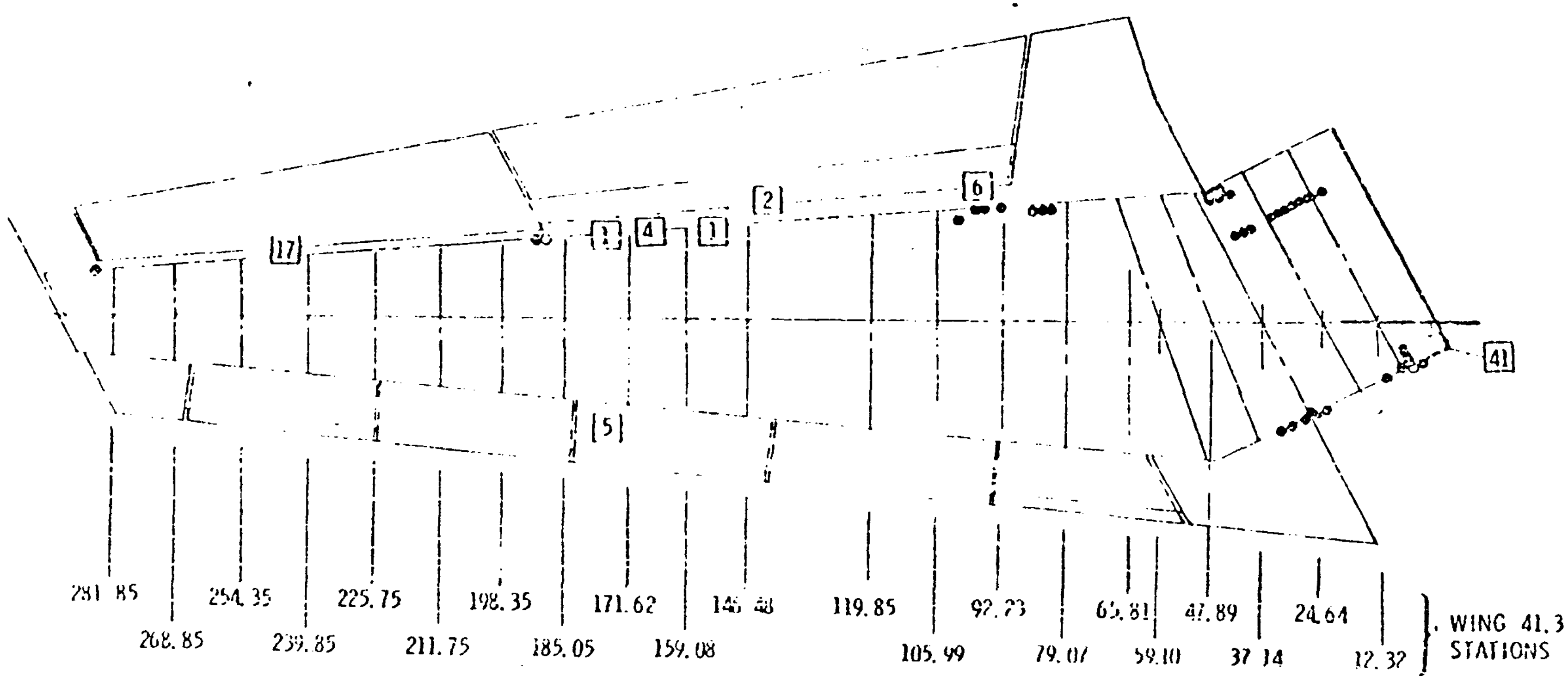


FIGURE 28 T-39A Right Wing Crack Distribution - Service & Test

3.3 BOMBER AIRCRAFT

The bomber aircraft discussed in this report are the B-52, FB-111A and F-111C.

3.3.1 B-52 Series

The B-52A and B aircraft design was initiated in 1951 and the B-52C-F were designed in the 1953-1954 time period. The B-52 is a 450,000 pound (B-52C-F) - 488,000 pound (B-52G-H) design gross weight heavy bomber class aircraft.

The original design was based on a high altitude mission requirement. In 1959 the B-52 mission was changed not only to include the low level mission but also to initiate the airborne alert mission. In 1962 the contour low level flying concept was initiated. All of those changes have had a marked effect upon design usage.

The B-52 was originally designed in accordance with the applicable requirements of the R-1803 series specifications. These specifications did not define design standards relative to service life expectancy, fatigue endurance, fatigue loads, fatigue testing much less damage-tolerant/fail safe design. However, fatigue requirements via aircraft design specifications have been implemented into improved modification programs incorporated on the B-52 fleet.

A review with ALC personnel on the durability problems associated with the B-52 service aircraft revealed the following:

1. L.E. Upper and Lower Chord - 7075-T6 spline cracks caused by impack wrench being used to remove steel fasteners in gap cover.

2. Rear Spar Web - 7075-T6 - Tooling hole in spar web caused shear failure.

3. Structural door failures - cracks generated from sloppy holes.

4. Trailing edge panels, flaps, leading edge panels cracked from excessive noise.

5. ECP 1581 Mod on first 30 airplanes was very bad due to poor jig design, inadequate training of people, and not enough attention given to design detail.

6. Fuel cell mod problems associated with mislocated rib, installation misfits and inexperienced new hires.

7. Extensive cracking occurred in the sculptured skins.

8. Service cracks experienced in upper longeron due to rework. Material changed from 7075-T6 to 7075-T73.

9. Cracking in leading edge ribs of horizontal stabilizer.

10. Corrosion is primary problem in horizontal stabilizer.

11. De-icing of runway is the single contributory cause of lower surface corrosion.

12. Mechanical problems with opening and closing the bomb-bay doors.

13. Body side skins replaced with like design and material one gage thicker. No problems with skins except aging.

14. Landing gear trunnion cracking caused by poor design. Undercut stress riser in small radius.

15. Few problems in MLG Bulkheads caused by radius cracking in T-sections due to clamping action and improperly shimmed installation.

16. Forging parting plane cracks precipitated by corrosion.

17. Problems in pressurized compartment center mostly around seals such as windows, doors, etc.

18. Catastrophic failure on one airplane due to stress riser in fastener hole of rabbet cut.

19. Body station (BS) 1655 bulkhead (original welded steel design) was replaced with a one piece forged steel crown with significantly strengthened lower members. This modification was the result of four B-52C-F accidents caused by failure of the BS 1655 bulkhead.

20. The center wing upper front spar chord on the B-52G/H aircraft had extensive machining which exposed end grain of the material thus affording extensive susceptibility to stress corrosion cracking. Cracks were noted on 10 aircraft during mod program.

B-52 Cyclic Fatigue Test Program

The need for a complete fatigue evaluation program for the B-52 airplane was made apparent by the B-47 airplane fatigue failures in 1958 and the similarity of the design criteria and usage requirements of the two airplanes (Reference B-52 ASIP)(Ref. 6).

The primary objectives of the B-52 fatigue evaluation program were:

1. To define fatigue areas
2. To determine adequacy of proposed repairs.
3. To establish required inspection procedures.

The results of four major B-52 series airframe test programs are discussed in the following paragraphs. Program results include:

1. B-52A-F Body, Wing, Fin and Stabilizer
2. B-52G Wing
3. B-52G/H Wing and Body (ECP 1050)
4. B-52G/H Body and Empennage (ECP 1128/ECP 1185)

B-52 A-F Cyclic Test

- o Fwd body test - 10,000 hours
  - No failures
- o Fwd body cyclic pressure test - 25,000 hours
  - 3 pressure web failures
  - 7 pressure floor to body frame tie angles.
- o Fwd body cyclic pressure test - B-52D service A/C 88,500 Equiv. Hours
  - Side skin cracks and pulled through rivets
  - Pilot and co-pilot escape hatch latch arm cracks
- o Fin test - 10,000 Equiv. Flt. Hours
  - Fin skin cracking
  - Fin terminal pin cracking
  - 1655 welded steel bulkhead
  - Aft body lower skin cracking
- o Wing test - 10,000 Equiv. Flt. Hours
  - o 52 major and 55 minor failures
    - Main wing skin @ rear spar
    - Upper and lower wing access doors
    - Wing upper surface skin splices
    - Wing lower surface closure panels
- o Wing Test Follow-on #1 - 5000 Equiv. Flt. Hours
  - o 22 Major and 36 minor failures
    - Wing closure panels
    - Fuel sump drains
    - Inspar wing skin trailing edge.

- o Wing Test Follow-on #2 - 5000 Equiv. Flt Hours
  - o 138 Major and 31 minor failures
    - Upper and lower wing skin panels
    - Lower wing stiffeners
    - Rear spar
    - Wing splice
    - Wing lower surface cutout doublers
- o Wing test follow-on #3 - 5000 Equiv. Flt. Hours
  - o 88 Major and 9 minor failures
    - Upper and lower wing skin panels
    - Wing lower stiffeners
    - Upper and lower wing splice areas
    - Rear spar web
- o Wing Test Follow-on #4 - 5000 Equiv. Flt. Hours
  - o 113 major and 3 minor failures
    - Lower wing skin panel
    - Wing lower stiffeners
    - Rear spar web
    - Body crown skin
- o Wing Test Follow-on #5 - 5000 (35,000 to total) Equiv. Flt. Hours
  - o 55 major and 1 minor failure
    - Lower wing skin panels
    - Upper wing skin panels
    - Upper wing splice plates
    - Rear spar web
    - Rear spar chord
    - Boomerang fitting
    - Lower wing stiffeners
    - Wing rib web
    - Body side skin
    - Upper longeron and bulkhead chord

- o Stabilizer Test - 10,000 Equiv. Flt. Hours
  - No significant failures
- o Stabilizer Test Follow-on - 15,000 Equiv. Flt. Hours
  - No significant failures

B-52G Wing Cyclic Test

10,992 - Equivalent Flight Hours

- Ten **critical areas identified**
  - Lower wing skin panel - 59 failures originated in rabbet cut in fastener holes or areas of discontinuity.
  - Lower wing skin panel - area of drag strut fitting and fairing clip fastener holes
  - Lower wing skin panel boost pump fitting areas - 31 failures
  - Lower wing skin panel fuel drain hole L/R
  - Upper aft wing skin panel rabbet cut
  - Lower wing panel stiffener sealant injection holes. Twenty-eight failures
  - Upper aft wing skin panel - 2 failures
  - Missile lower fittings - 3 failures
  - Lower surface access door area - 8 failures
  - Lower wing skin panel stiffeners at wing rib intersection attachment holes.

B-52G/H (ECP 1050) Wing and Body Cyclic Test

Flt x Flt Spectrum - 4 Life Times, 48,560 Equiv. Flt. Hours

- o 1304 Combat crew training missions
- o 1480 Extended high altitude missions

o A total of 2132 test specimen defects varying in magnitude and origin were discovered during test. Significant failures include:

- Center wing rib web cracks
- Lower wing stiffener inspan cracks
- Body fatigue cracking in channel beam tie to fuel deck and BS 805 Bulkhead
- Numerous body cracks in upper skin fastener holes common to longeron
- Separate cracks in upper longerons
- Body shear skin panel cracks
- Drag angle fatigue cracks
- Crown skin cracking in fastener holes
- Trailing edge wing structure cracks

o Follow on Testing - 4.0 to 4.65 Lifetimes, 4.65 to 5.30 Lifetimes and 5.30 to 6.0 Lifetimes

- Cracks in upper chord of front spar
- Major crack in L/H upper longeron fitting and smaller crack in R/H fitting
- Rear spar web cracks L/R
- Rear spar lower chords
- Front spar span wise cracking
- BS 538 Bulkhead forbing forward skin attach flange crack
- Corner tie fittings common to front spar and inspar ribs crack (6)
- Lower wing skin stiffeners (18 cracks)
- Rear spar lower chord crack
- Boomerang fitting fatigue crack
- L/R Drag angle fatigue cracks
- Lower trailing edge shelf panel seal beam structure cracks (19 incidents)
- Extensive galling occurred between pin and bushings @ L/H aft terminal pin location.

B-52G/H (ECP 1128/ECP 1185)
Body and Empennage Cyclic Test

- o Flt x Flt Spectrum - 4 Lifetimes
  - o 1720 Combat Crew Training Missions
  - o 420 Combat Crew Supplemental Missions
  - o 196 Airborne Alert Indoctrination Missions
  - o 96 Airborne Alert Indoctrination Supplemental Missions
  - o 16 Combat Crew Training Proof Missions

- o A total of 688 discrepancies of various types, magnitudes and origins were discovered during test. These include the following:
  - Fatigue cracks in R/H wing primary structure near inboard and outboard nacelle area.
  - Fatigue crack in flap trap rib
  - Numerous cracks (385) in the inboard trailing edge assemblies. Cracks originated in hat sections and trailing edge upper skin surface
  - Cracks in L/R drag angles
  - Fatigue skin cracks around wheel well caused by severe canning
  - Twenty-two (22) cracks discovered on the horizontal stabilizer. Most cracking occurred on lower stabilizer spider fitting.

FOLLOW-ON TESTING (4.0 - 7.6 LIFETIMES)

- o 292 Airborne Alert Indoctrination Missions
- o 2140 Combat Crew Training Standard Missions
- o 16 Combat Crew Training Missions
- o 6400 Constant Amplitude cycles

- A total of 270 discrepancies were discovered during the follow-on testing. Among the more significant failures are the following:

  - Several major fatigue cracks occurred in the inspar structure; namely, rear and front spar web cracks and upper chord front spar

  - Upper wing panel cracked in fuel filler hole

  - Drag angle and boomerang fitting cracks

  - Numerous cracks in the inboard section of the trailing edge assemblies

  - Body damage in tank deck beam angles

  - Additional wheel well skin and body skin cracking

  - Twenty-eight fastener failures in bomb bay door hinge area

Numerous ECPs have been incorporated into the B-52 fleet to improve the strength and fatigue resistance of the body, wing and empennage based on cyclic test results, fleet experience and SAC planned usage. An example of these ECP type changes is illustrated in Figure 29. They include such modifications as:

- New wing skins - 2024-T351
- New upper surface inspar panels with increased thickness and 7075-T6 material
- Eliminate discontinuities
- Lower stress levels in joints and splices
- Use of taper shank fasteners for high strength attachment
- Deletion of fuel drain holes
- Use of taper shank fasteners to attach brackets and clamps to wing skins, spars and stiffeners
- Apply new protective coatings to internal structure of wing fuel tanks
- Replace existing fasteners with oversize fasteners in certain areas

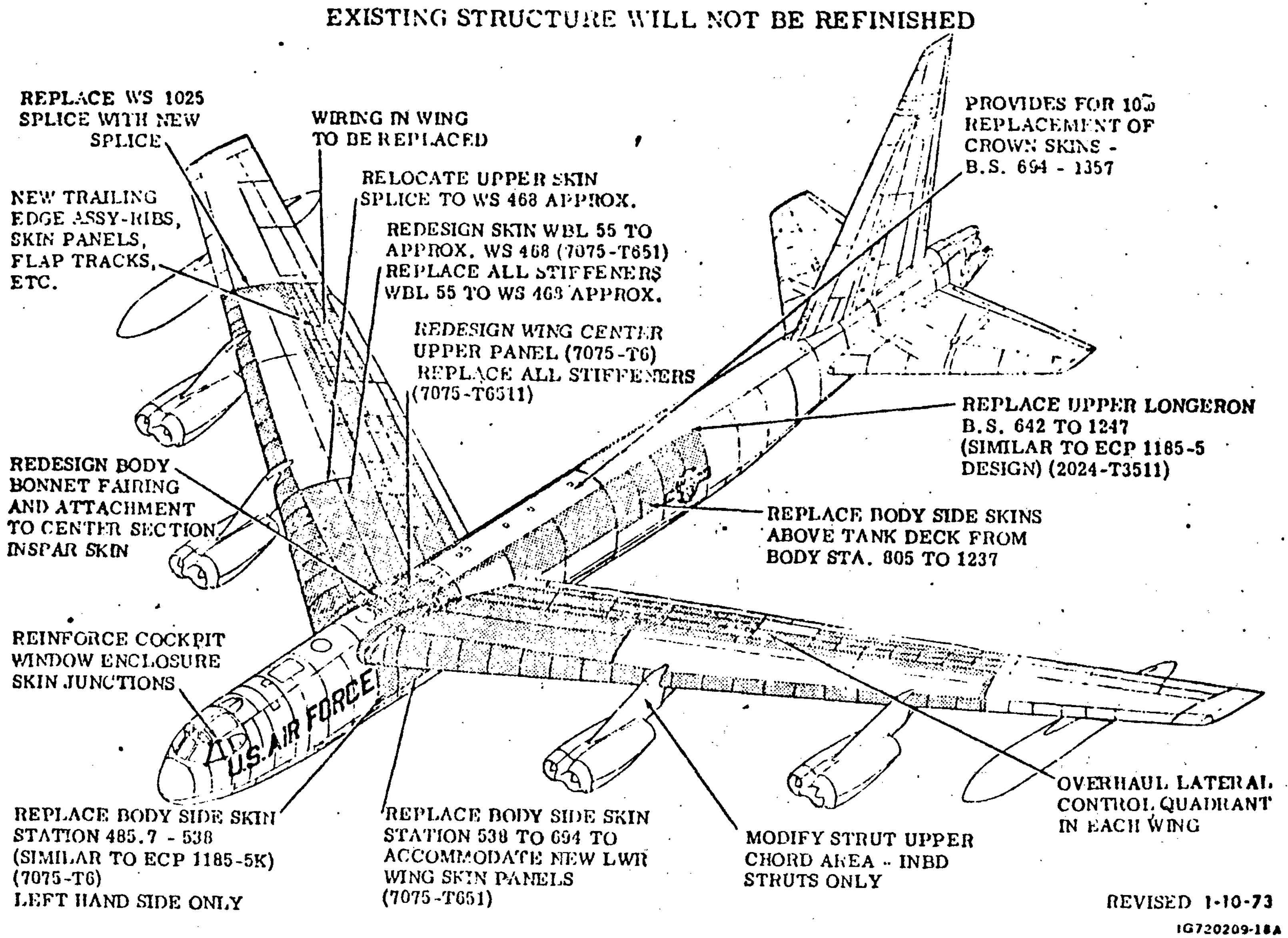


FIGURE 29 B-52 ECP 1581

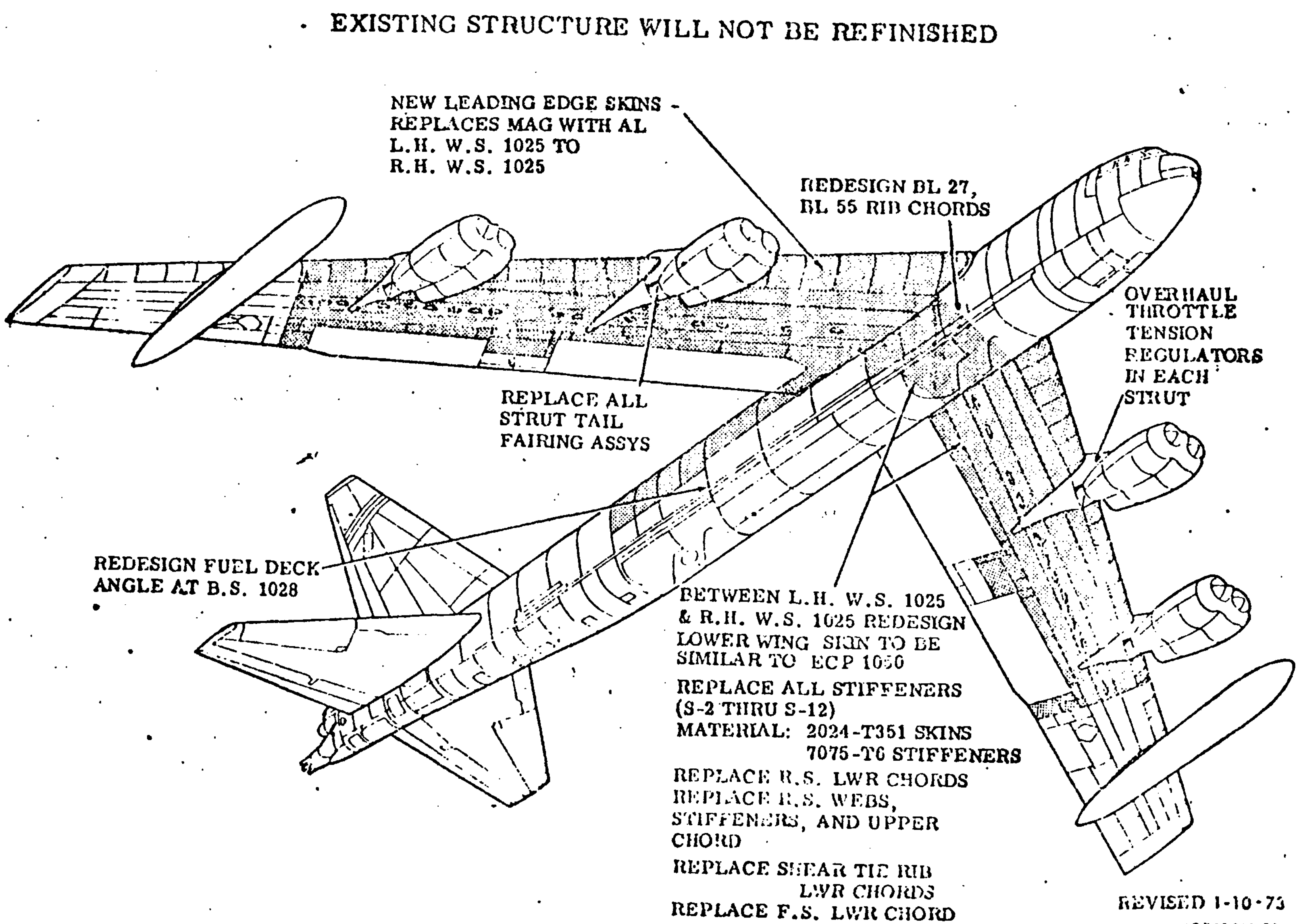


FIGURE 29 B-52 ECP 1581 (Cont'd)

3.3.2 FB-111A Fighter/Bomber

The FB-111A is a Class 2 bomber version of the F-111A with SRAM capability. It is equipped with TF30-P-7 engines, TP II inlet, increased weight capability landing gear, and longer wing span (3.5 foot increase per wing).

The FB-111A test goal was for 10,000 hours of safe service life which was accomplished by the application of 40,000 equivalent flight hours of spectrum loadings derived from bomber usage and FB-111A mission profiles.

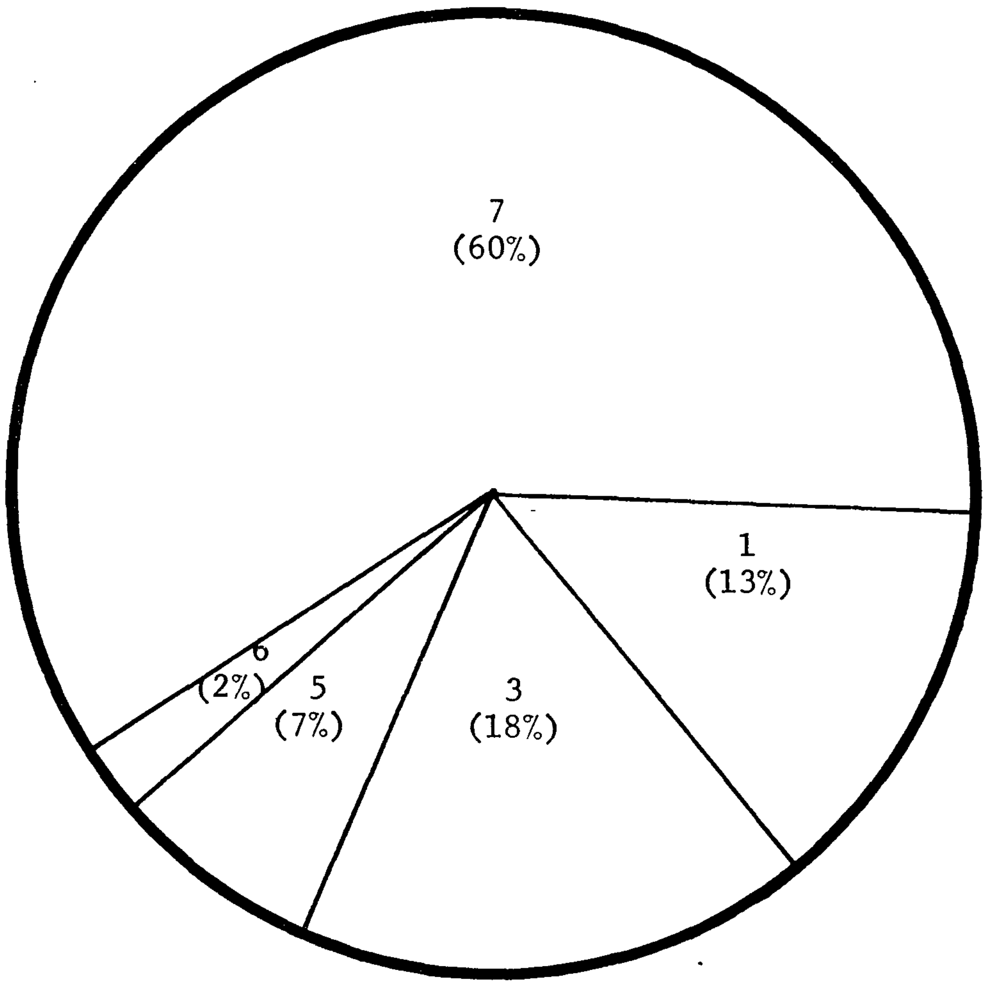
During the cyclic test program there were 45 recorded test incidents. A breakdown of these results is shown in Figure 30.

Two significant service failures experienced on the FB-111A aircraft have been:

1. Speed brake column/truss on the door. The fatigue failure originated in the lugs of both the column and truss. These same areas experienced test failures.

2. Upper plate on the wing carry-thru box. Crack found during cold proof test. The fatigue failure originated in a sealant groove hole. Examination revealed a bit had been broken off in the hole and had subsequently been eloxed out.

Additional durability problems experienced on the FB-111A are described in the Analytical Condition Inspection (ACI) results shown in Table 4 on page 50.



1 6 Incidents of Broken Fasteners

2 0 Receptacle Failure

3 8 Cracking Incidents

4 0 Panel Delamination

5 3 Shim Failures

6 1 Incident of Gouge, Dent, Scratch

7 27 Incidents of Galling, Wear, Hole Non-Roundness

45 TEST INCIDENTS
(40000 Equivalent Flight Hours)

FIGURE 30 FB-111 Airframe Test Program Summary

3.3.3 F-111C Fighter/Bomber

The F-111C is a version of the F-111A designated for the Royal Australian Air Force (RAAF). It has the FB-111A long wings, increased weight capability landing gear and the F-111 improved carry-thru-box structure.

Service data obtained from the Royal Australian Air Force on the F-111C aircraft were in the form of Analytical Condition Inspection data. These data include ACI findings on a set of wings (F-111C A8-135) the results of which are shown in Table 7 and the findings from four aircraft the results of which are shown in Tables 8 through 11 and summarized in Table 12.

It is observed that the defects generally fall into three main categories; namely cracking, corrosion, and maintenance/manufacturing defects.

Cracking has been found in some brackets and flanges but they have not been structurally significant. Wing pivot fitting bushings have been found to be cracked on most aircraft. This is an identical problem experienced by the USAF. This problem does not have a detrimental affect on aircraft operation. It is only considered a maintenance problem.

The main areas affected by corrosion have been the Wing Pivot Fitting and Horizontal Stabilizer bearing housing. In some cases the corrosion has been fairly widespread but not deep enough to have any structural implications.

Manufacturing defects found have been due mainly to poor machining resulting in grindouts, incorrect spot facing, gouges, and machine tool marks. The most significant areas were found in the WPF fuel flow doors which were corrected by smoothing out depressions and radiusing sharp corners.

TABLE 7 F-111C ANALYTICAL CONDITION INSPECTION (A-8-135 WINGS)

INSPECTION ARTICLE	STRUCTURE DESCRIPTION	ACI FINDINGS
F-111C A-8-135 WINGS	WINGS-EXTERNAL	L/H - MINOR PAINT DETERIORATION: OTHERWISE UNBLEMISHED. R/H - LOWER SURFACE EXPERIENCED MINOR CORROSION ALONG ENTIRE LENGTH OF WING. TWO TAPER-LOK BOLTS PROTRUDED FROM LOWER SURFACE. NO DETERIORATION OR DELAMINATION FOUND ON LOWER SURFACE BORON DOUBLER.
	PIVOTS	WING PIVOT BUSHINGS CRACKED BEYOND REPAIRABLE LIMITS: COMMON PROBLEM CAUSED BY HIGH RADIAL COMPRESSIVE STRESSES DUE TO FLIGHT BENDING MOMENTS
	FUEL TANKS AND ASSOCIATED SYSTEMS	MINOR FUEL LEAKS AND MINOR PAINT DETERIORATION - PLUMBING IN GOOD CONDITION.
	FLAPS	IN GENERAL - POOR CONDITION: PAINT DETERIORATION, MINOR CORROSION, MINOR DISBONDS, SEALANT REVERSION ALONG TE. DAMAGED ROLLERS ON FLAP CARRIAGE ASSEMBLIES.
	SPOILERS	INBOARD SPOILER ON R/H WING MISSING ALONG WITH BOTH ACTUATORS. RIGHT WING BELLCRANK REWORK REQUIRED TO REDUCE STIFFNESS.
	SLATS	CORROSION AND PAINT DETERIORATION, MINOR DENTS, SLAT TRACK ROLLERS DAMAGED.
	WING LEADING EDGE	GOOD CONDITION.

TABLE 8 F-111C ACI SUMMARY (AIRCRAFT NO. A8-127)

Item	Condition	Defect Report
Beam 12K3213	Sealant Reversion on LH Side of Beam Around Screen Anchor Nuts.	
Fairings 12B8223-5 and 12B8223-6	Removal of ECM Antenna Covers Revealed Cracks in Fairings.	3AD/13/75
Upper Translating Cowl Rail Attachment Brackets	Corroded.	3AD/74/74 RH 3AD/75/74 LH
Bracket Attachment 12B7725-1	Cracked.	3AD/54/74
Bulkhead 12B2104	Production Repair Discovered - Crack Stop Drilled and Scab Patched - Repair Functional but Workmanship Sub-Standard.	
Bulkhead 12B2615	Crack Discovered - Prev Stop Drilled.	3AD/72/74
	ACI SEGMENT 5 A8-127	
Strake 12B13406	Delamanations in Leading Edge.	
FWD Engine Access Door	Hinge Attachment Bolts Sheared.	3AD/70/74
Saddle Tank Structure	Six (6) Rivets in RH Saddle Tank have Incomplete Tails.	
Longeron 12B10812	Left and Right Longerons have Superficial Damage Between FS632-670--Caused by Drilling Holes for Anchor Nuts in Heat Shield Supports.	
Lower Longeron Splice 12B10503	Alum Metal Spray Coating Over D6AC Steel Structure Missing.	3AD/73/74
Panel 3325	Large Dent on Inside Surface.	3AD/80/74
Splice Plate 12B10618	Fasteners Missing Washers, One Bolt has Gap of .038 Between Nut and Frame.	3AD/57/74 3AD/58/74
	ACI SEGMENT 2 A8-132	
Radome Latches	Minor Sealant Reversion.	
Radome Latch Assy	Loose Rivets in Assy Cover.	

TABLE 8 F-111C ACI SUMMARY (AIRCRAFT NO. A8-127) (Cont'd)

Item	Condition	D/R No
Mainplanes		
Wing Pivot Fitting	Surface Corrosion	3AD/59/74
Wing Pivot Fitting	Surface Corrosion	3AD/60/74
Plate WDF	Faulty Machining	3AD/63/74
Plate WDF	Faulty Machining	3AD/64/74
WDF Centre Spar Flange	Depression (Grindout)	3AD/97/74
Wing Pivot Fitting	Pin Head Blow Hole	3AD/119/74
Slat Assy No 4	Void Area	3AD/66/74
Slat Assy No 1	Loose Rivets Attache Ice Scraper	3AD/67/74
WDF Bush	Cracked and Corroded	3AD/24/75
Wing Assy	Sealant Reversion	3AD/71/74
Housing Slat Track	Broken Flange	3AD/139/75
Fuselage Rear		
Panel Lower 12B10102-3	Large Depression	3AD/89/74
Seal Press 12B10383-10	Sealant Reversion - Cracked	3AD/98/74
Fuselage Centre		
WCTB Support Brackets 12B12351-13/-14	Ser Nos Etched in Bracket	3AD/114/74
WCTB Shear Panel	Scratches and Gouges in Fuel Fuel Inter Conn Attach Holes	3AD/141/74
RAM Air Scoop	Cracked	3AD/206/75
Panel Water Tank	Cracked Flange	3AD/92/74
Skin Water Tank	Cracked and Corroded	3AD/93/74
Skin Water Tank (New)	Pivot Holes Misaligned	3AD/94/74
Skin Water Tank (New)	Poor Application Sealant	3AD/5/74
Bolt AIC 260E-9-34	Corroded by Water Ingestion	3AD/121/74
Trans Cowl Beam Assy	Prev Repair O/S Holes	3AD/10/75

TABLE 8 F-111C ACI SUMMARY (AIRCRAFT NO. A8-127) (Cont'd)

Item	Condition	D/R No
<u>Air Inlet Area</u>		
Top Hat Stiffeners	Cracked	3AD/68/74
Skin Support Prim Inlet	O/S Holes Tran Cowl Track	3AD/117/74
Stiffener 12P11650-60	Mismatch Prev Repair	3AD/122/74
<u>Empennage</u>		
Bulkhead (STA 786.5)	Fin Attach Bolt Hole Off Centre and Bushed - Prev Repair	3AD/115/75
Horiz Stabilizer LH	Corrosion in Brg Hsg	3AD/11/75
<u>Hydraulics</u>		
Support Bracket Piping	Cracked	3AD/53/74
<u>Air Conditioning</u>		
Duct Air 12Y838-809	Cracked	3AD/39/75
<u>Canopies</u>		
RH Windshield	Sealant Reversion	3AD/82/74

TABLE 9 F-111C ACI SUMMARY (AIRCRAFT NO. A8-130)

Item	Condition	D/R No
Mainplanes		
Pivot Pin Bushes	Cracked	3AD/75/75
Wing Pivot Fitting LH	Corrosion	3AD/103/75
Wing Pivot Fitting RH	Corr on Lower Plate Adj Boron Doubler	3AD/125/75
Lever Assy Pivot Pylon	Corrosion	3AD/83/75
Wing Assy Lower Skin	Taperloc Protrusion	3AD/135/75
Plate WDF LH and RH	Machining Mark - Spot Facings	3AD/165/75
Wing Pivot Fitting RH	Corrosion and Discoloration	3AD/171/75
Wing Assy RH	Taperloc Protrusion	3AD/168/75
Fuselage Rear		
Door Fwd Eng Access LH	Water Ingress	3AD/185/75
Splice Plate	Cracked	3AD/177/75
Splice Plate	Deep Spot Face	3AD/182/75
Fuselage Centre		
Overwing Fairing	Cracked Self Locking Nut	3AD/113/75
Skin Water Tank	Cracked	3AD/138/75
Empennage		
Horizontal Stab	Corrosion in Brg Hsg	3AD/96/75
Horizontal Stab	Corrosion in Brg Hsg	3AD/97/75
Vertical Stab	Bowing in Leading Edge	3AD/86/75
Horizontal Stab	Pivot Shaft Bushings Scored	3AD/152/75
Vertical Stab	Sealant Reversion	3AD/178/75

TABLE 10 F-111C ACI SUMMARY (AIRCRAFT NO. A8-132)

Item	Condition	D/R No
Mainplanes		
WCTB Pivot Brgs	Corroded	3AD/207/75
LH Wing Upper Skin	Delamination	3AD/4/76
Fuselage Rear		
Splice Plate	Cracked	3AD/5/76
Fuselage Centre		
RAM Air Scoop	Cracked	3AD/206/75
Skin Water Tank	Cracked	3AD/208/75
Empennage		
Horizontal Stab	Corrosion in Brg Hsg	3AD/204/75
Horizontal Stab	Corrosion in Brg Hsg	3AD/205/75

TABLE 11 F-111C ACI SUMMARY (AIRCRAFT NO. A8-133)

Item	Condition	D/R No
Mainplanes		
Pivot Pin Bushes	Cracked	3AD/120/74
Pivot Pin Retainer Hole	Two Depressions (Grindouts)	3AD/34/75
Centre Spar Flange	Machining Marks	3AD/43/75
Centre Spar Flange	Grindouts and Shim Fitment	3AD/57/75
Pivot Pin Bushing	Corrosion	3AD/85/75
Plate WDF LH	Machine Marks - Spot Facings	3AD/163/75
Plate WDF RH	Machine Marks - Spot Facings	3AD/164/75
Fuselage Centre		
RAM Air Scoop LH	Loose Attachment	3AD/76/75
Splice Plate 12B7933	Rough Machining	3AD/146/75
Skin Assy - Water Tank	Cracked in Lower Flange	3AD/137/75
Air Inlet Area		
Engine Inlet Duct	Corrosion around Rivet Heads	3AD/173/75
Empennage		
Horizontal Stab	Mismatch - New Tip	3AD/58/75
Horizontal Stab LH	Corrosion in Brg Housings	3AD/77/75
Horizontal Stab RH	Corrosion in Brg Housings	3AD/78/75
Horizontal Pivot Shaft	Bushing Scored	3AD/15/75
Undercarriage		
MLG Retract Act Bracket	Previous Rework of Gouge	3AD/41/75
MLG Lateral Beam	Chafing	3AD/56/75
MLG Support Beam Assy	Bushing Wear	3AD/114/75
MLG Support Beam Assy	Bushing Wear	3AD/115/75
MLG Lower Shock Pin	Bushes Corroded	3AD/116/75

TABLE 11 F-111C ACI SUMMARY (AIRCRAFT NO. A8-133) (Cont'd)

Item	Condition	Defect Report
Shear Panel 12B3803	Deep Score on Panel-Boundary Layer Air Scoop Cracked.	3AD/35/75
Panel 12B3681	Medium Surface Corrosion, Small Dent, No Delamination.	
Panel 12B3682	Slight Scratch Marks.	
Nacelle Formers 12B2907, 12B2908 12B2909	12B2907 - Light Corrosion, Area Chafed by Cracked and Loose Ram Air Scoop. 12B2908 - Light Surface Corrosion. 12B2909 - Light Surface Corrosion.	3AD/40/75
Longeron 12B1311	Corrosion on Aft End of Longeron (LH) and Fasteners.	
Fuel Decks 12B4151, 12B4117	Minor Dents, No Delaminations.	
Longeron 12B1904	Grinder Damage Around the Top of Bolt Holes.	
Panel 3111	Dents on Panel, No Delaminations.	
	<u>ACI SEGMENT 7 A8-130</u>	
Frame 12B13187 FS799.00	Corrosion.	
Upper Longeron Aft Centre Body 12B13118	Corrosion. Also Found on A8-133, A8-132	3AD/107/75 3AD/158/75-189/75
Horiz Stab Pivot Bulkhead	Corrosion in Internal Bore of Pistol Fitting (Also Found on A8-133)	3AD/111/75 3AD/112/75
Vertical Stab	Leading Edge Bowed	3AD/86/75

TABLE 12 F-111C ACI SUMMARY (4 AIRCRAFT)

AIRCRAFT DESIGNATION	FLIGHT HOURS @ ACI
A8-127	233
A8-130	392
A8-132	621
A8-133	426

DESCRIPTION OF DAMAGE	CUMULATIVE OCCURRENCES
o CRACKING	23
o CORROSION	27
o FASTENER RELATED - missing/loose	8
o DENTS/NICKS/SCRATCHES	7
o DELAMINATION	2
o WEAR	4
o MAINTENANCE/MANUFACTURING - repairs, machine marks, grindouts shim, mismatch, etc.	23
o OTHER - sealant reversion	9

3.4 CARGO/TRANSPORT AIRCRAFT

3.4.1 C-130 Series

The C-130 airplane is a turboprop transport designed and built for the U.S. Air Force.

There are several basic models of the C-130 which include the C-130A, C-130B, C-130E and C-130H models. Several variations of each of the basic models have been built and are used in a variety of different missions.

The C-130 aircraft serve the Material Airlift Command (MAC) and the Tactical Air Command (TAC).

The C-130 fatigue test program identified fatigue induced areas in which corrective action was accomplished prior to fleet degradation. Distribution of major test failures among the C-130 aircraft is shown in Table 13.

TABLE 13

C-130 FATIGUE TEST SUMMARY

SPECIMEN	MAJOR FAILURES	TEST LIFETIMES @ FAILURE
C-130A Fuselage	2	1.3
C-130B Empennage	3	8.4
C-130B Wing	3	2.0
C-130E Wing		
- Center	18	2.3
- Outer	9	3.0
C-130B/E Wing		
- Center	4	4.4
- Outer	9	4.4

Fatigue sensitive areas of the C-130 aircraft are identified in Figures 31 , 32 and 33 for the fuselage, center and outer wing.

Some of the C-130 service failures identified to date are shown in Table 14.

ITEM NO.	AREA DESCRIPTION	MDS
1.	WINDSHIELD POST	C-130A/B/E/HCH/P/N
2.	CREW DOOR LONGERON	C-130A/B/E/HCH/P/N
3.	FORWARD WING TO - FUSELAGE FAIRING FS 477	C-130A/B/E/HCH/P/N
4.	FS 597 PORK CHOP - FITTING	C-130A/B/E/HCH/P/N
5.	FS 597 VERTICAL BEAM	C-130A/B/E/HCH/P/N
6.	BOW BEAM END FITTING	C-130A/B/E/HCH/P/N

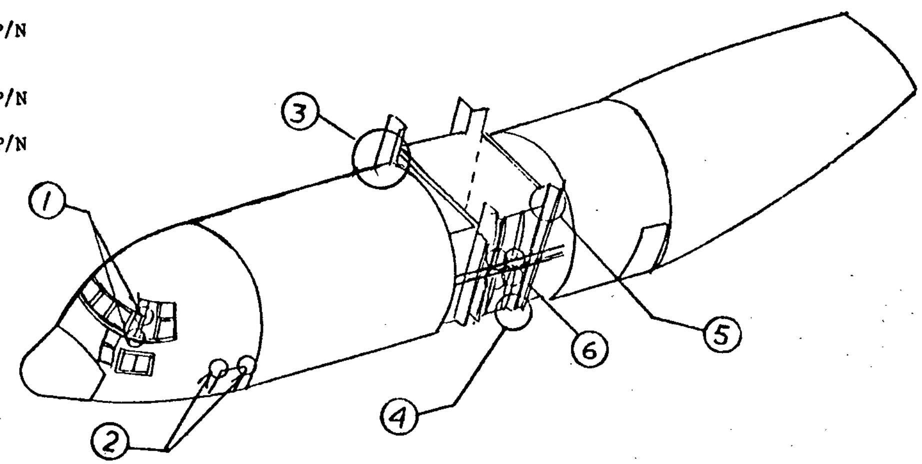
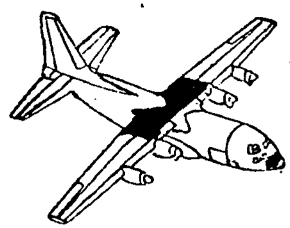


FIGURE 31 C-130 Fuselage Fatigue Sensitive Areas



ITEM NO.	AREA DESCRIPTION	MDS
1.	UPPER FRONT BEAM CAP - CWS 55-60	A
2.	UPPER REAR BEAM CAP CWS 20 & 55-60	A
3.	RANDOM LOWER BOX BEAM - SKIN CRACKS	A/B/E HC-130HCH/P/N C-130H

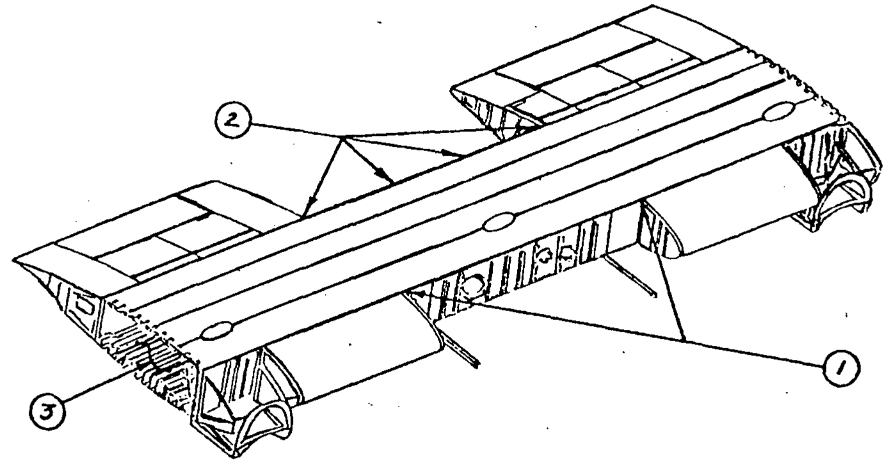


FIGURE 32 C-130 Center Wing Fatigue Sensitive Areas

ITEM NO.	AREA DESCRIPTION	MDS
1.	LOWER FRONT BEAM CAP - OWS 162/197	C-130B/E/HCH/P/N
2.	LOWER FRONT BEAM CAP - OWS 54/108	C-130E/HCH/P/N
3.	LOWER REAR BEAM CAP - OWS 54/108	C-130E/HCH/P/N
4.	FRONT BEAM WEB - OWS 158/201	C-130B/E/HCH/P/N
5.	FRONT BEAM WEB OWS 70	C-130E/HCH/P/N
6.	LOWER SKIN PANEL - OWS 162/197	C-130B/E/HCH/P/N
7.	LOWER SKIN PANEL 35	C-130B/E/HCH/P/N
8.	LOWER SKIN PANEL OWS 6	C-130B/E/HCH/P/N

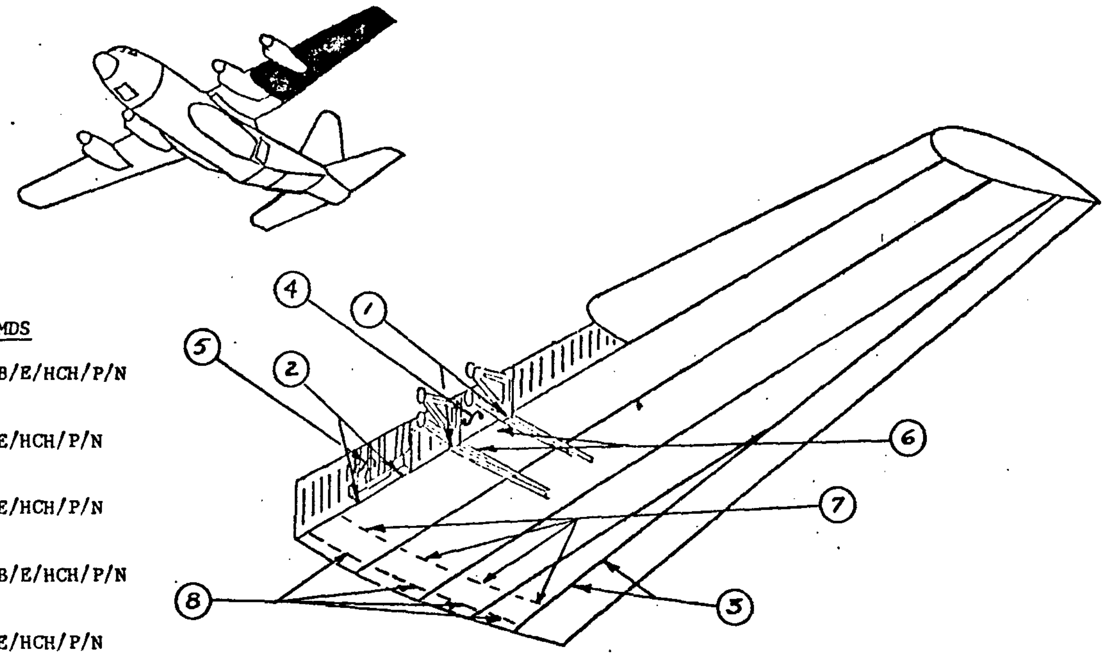


FIGURE 33 C-130 Outer Wing Fatigue Sensitive Areas

TABLE 14 C-130 SERVICE FAILURES

PART IDENTIFICATION	NO. OF ACFT	FAILURE ORIGIN
o Lower Front Beam Cap - OWS 158	29	Sharp radii in vertical and horizontal flange
o Front Beam Web Crack - OWS 162/197	155	Radius crack
o Lower Skin Panel - OWS 162/197	10	Fastener hole crack caused by local overload
o Lower Skin Panel - OWS 35	30	Doubler - Botch-up hole
o Lower Beam Cap - OWS 54 & 108	33	Poor quality hole and short edge distance on cap
o Lower Skin Panel - OWS 6	1	Fastener hole - corrosion and fatigue
o Engine Truss Mount - Sway Brace Lug - Wing Attach Lug	72	7075-T6 Forging Lug - Corrosion

Other durability problems identified with the C-130 aircraft are listed below.

- C-130A - Old Center Wing

  - Fatigue cracking originating in a cutout of an upper surface spar cap loaded by Taxi and Assault Landing

  - Fatigue cracking originating in a nutplate rivet hole of the CW station 60 front spar (7075-T6)

- C-130A - New Center Wing

  - Clean design with material change from 7075-T6 to 7075-T73 and rectangular cutouts changed to elliptical.

- C-130 Outer Wing

  - Current most critical component

  - Lower rear box beam spar cap has short edge distance. Failures originate in fastener hole.

  - Corrosion in lower wing skin in fuel area.

  - Corrosion cracking also in dry area due to fumes and moisture condensation

- MLG Fuselage Area

  - Failures in flange cutout of stretch bow beam (7075-T6)

- Fuselage Area

  - Cracking in Pork Chop Fitting (7075-T6 forging). Crack originated in a radius loaded by higher than predicted fuselage deflection. Material changed to steel for "H" model.

3.4.2 C-141A Aircraft

The following discussion has been extracted from the Reference 7 paper. This paper received the Best Paper Award at the 9th Annual SAMPE Technical Conference held in Atlanta, Ga, October 1977.

The C-141 is a high performance, long range heavy logistics transport. The aircraft was designed to the MIL-A-8860 specifications during the 1963 time frame utilizing the technology of the late 1950s and early 1960s. About 85 percent of the aircraft utilization is logistics and the other 15 percent is training.

The primary structure of the C-141A is a combination of high strength aluminum and steel alloys; namely 7075-T6, 7079-T6 and 4340 steel.

Fatigue cracking on the C-141A structure has not been a major problem. This is principally the result of the low to moderate operating stress levels for most of the C-141A structure.

Another favorable experience is the absence of corrosion in the wing fuel tanks. This can be attributed to the corrosion preventative protection initially given during manufacture. This protection consisted of applying to the 7075-T6 surface a MIL-A-8625, Type II sulfuric acid anodize and MIL-C-27725 polyurethane topcoat. Faying surfaces were sealed with MIL-A-8802, and fasteners were installed wet with the same material.

Fuel leakage has only been a small problem and has occurred where straight shank fasteners have been used and around fasteners which were installed in larger than allowable holes. Fuel leaks around Tapor-Lok fasteners has not been a problem.

Although corrosion has not been a problem in fuel tank areas, it has been a <u>major</u> C-141A problem on much of the 7075-T6 structure where adequate corrosion protection was not applied.

o Upper Surface Wing Panels

The integrally stiffened wing panels of the C-141A aircraft were manufactured from 7075-T6511 extrusion. Steel Taper-Lok fasteners are used to attach the panels to one another and to various internal structure. The original finish system consisted of a MIL-C-8514 wash primer, followed by one coat of MIL-P-7962 primer and two coats of MIL-L-19537 acrylic lacquer. In time, this system cracked around the heads of the countersunk steel Taper-Lok fasteners permitting water entrapment and subsequent corrosion as depicted in Figure 34.

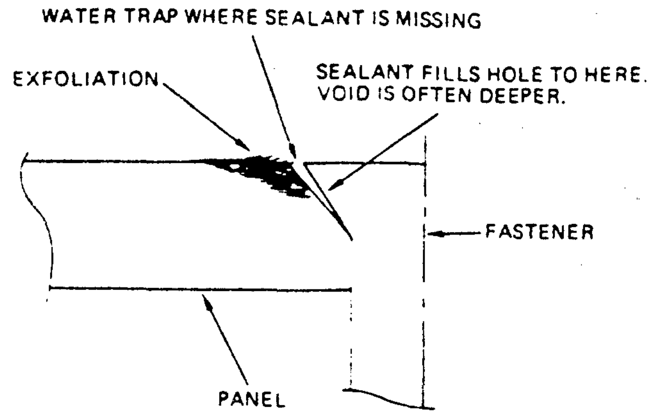


FIGURE 34 Corrosion Development Around Fasteners

Solution to this problem was the application of two coats of MIL-S-81733, Type III polysulfide primer followed by two coats of MIL-C-83286 polyurethane.

o Fuselage Main Frames

In the C-141A there are two main frames which transfer loads between wing and fuselage. Both are made from 7075-T6 forgings. During manufacture, these forgings were machined extensively, thereby exposing end grain which again resulted in stress corrosion cracking. Cracks have also originated from fastener holes, in the frame webs and along the various radii of the frame flanges. Solution to this problem is still being evaluated.

o Center Wing Panels

These panels have integral stiffeners machined from 7075-T6511 extrusions. The extrusions were hot joggled on each end at the chordwise splice between the center and inner wing. Stress corrosion cracks occur in the vertical leg of the integral stiffener, usually at the joggle as shown in Figure 35.

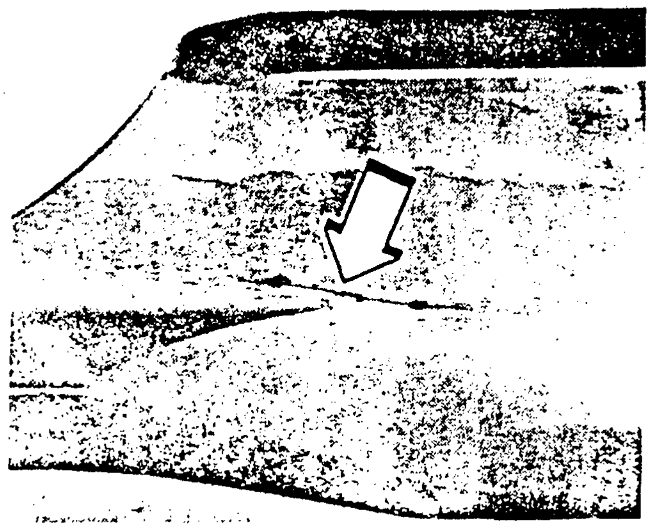


FIGURE 35 Crack in Stiffner at Joggle.

Metallurgical examinations of several crack stiffeners revealed that corrosions pits were forming in the time period between machining and anodizing thus significantly contributing to the subsequent corrosion cracking. This observation pointed to the fact that corrosion cannot be ignored during any part of material processing.

o 7079-T6 Aluminum Alloy

This material was also used extensively in the C-141A aircraft. Most of the fuselage skin panels are 7079-T6, the internal structure of the vertical and horizontal tail being 7079-T6 plate and forgings and the aluminum components of the landing gear being 7079-T6 forgings. Like the 7075-T6 material, corrosion cracking has also been a significant problem with the 7079-T6 material. A substitute material of 7049-T73 has been the apparent solution to the corrosion cracking problem with the landing gears.

The fuselage skins experienced the same corrosion cracking problem as discussed earlier with the upper surface wing panels. This problem has been significantly reduced with the use of MIL-S-81733, Type III polysulfide primer followed by two coats of MIL-C-83286 polyurethane.

o 4340 Steel

As initially delivered, most of the high strength steel in the C-141A was 260-280 KSI. Problems with this material have been due to stress corrosion and have occurred in landing gear components and wing and pylon mounted fittings. In the case of landing gear components, new cylinders made from 300M steel 260-280 KSI are being retrofited on the C-141A fleet.

o Pylon to Wing Attach System

The pylon attach fittings are quite susceptible to water intrusion. Although sealing this area is a requirement, accomplishment of this task is almost impossible due to the limited accessibility. Consequently, the environment is ideal for corrosion which is very common on these fittings. In cases where sealant is properly applied, corrosion does not exist; therefore, the problem becomes not one of sealant but use of a highly vulnerable material used in a limited access area bound to trap corrosive fluids. The significance of this problem cannot be over emphasized. The life cycle maintenance cost for the C-141A pylon attachment system is estimated to be in excess of 100 million 1976 dollars.

An example of this type corrosion on the C-141A is shown in Figure 36.

FIGURE 36 C-141A Pylon Attachment Fitting Corrosion.

The C-141A structural design makes extensive use of adhesively bonded components in secondary airframe structure and in several areas of primary structure in the fuselage pressure vessel. Bonded secondary structure is utilized in control surfaces, aerodynamic flow surfaces, and in general structural applications.

The original C-141A parts were assembled using the adhesive, surface preparation, and corrosive inhibitive technology of the late 50's and early 60's when structural bonding was in its development stages. Although the original polyamide epoxy film used displayed good mechanical properties during initial short term environmental tests, environmental susceptibility over the long term has proved to be a major problem. The primary factor has been moisture penetration into the bondline at the face sheet and adhesive film interface, resulting in adhesive failure because of desorption of the adhesive by water. The subsequent results were panel delamination and corrosion of the metallic structure which seriously degraded the structural integrity of the part.

During initial production, corrosion resistant honeycomb core and corrosive inhibitive bonding primers were not available. Panel edges, cutouts, and fastener holes were not sealed on installation, thus exposing the bondline to the environment. Panel faying and surface sealing of joints were accomplished in later production and during repair; however, maintenance costs for bonded honeycomb structure on the C-141A is astounding.

A recent major procurement, to supply replacement parts and spares for the original honeycomb components that have reached the end of their service life, cost approximately nine million dollars. At depot repair level where selected C-141A honeycomb parts are completely refurbished, 1977 expenditures approached 4.5 million dollars. This does not include field level daily support costs.

o Conclusions - C-141A Experience

Corrosion is currently the major C-141A material/process problem. It is prevalent in all forms including stress corrosion cracking, wherever 7075-T6 and 7079-T6 aluminum and 4340 steel are exposed to a corrosive environment. Second only to corrosion on the C-141A is the honeycomb problem.

These problems primarily stem from the materials and processes that were state-of-the-art during manufacture of the C-141A. Recent experience and current technology can significantly reduce these C-141 type problems on a new design.

3.4.3 KC-135 Aircraft

The KC-135 aircraft was an outgrowth of the Boeing developed jet transport prototype designated 367-80. It was designed to refuel the higher speed B-52. Production began in 1954 and five configurations of the basic aircraft were delivered before production was terminated in 1965. A total of 26 different designations of the -135 aircraft now exist.

Service life requirements were not specified during the initial design. Ten thousand hours were accepted as a desired life and design considerations indicated the airplane would be good to at least this level. Safe service lives have subsequently been established.

The following is a chronological history of significant events relating to durability of the KC-135 aircraft (Reference 8).

1962 through 1974 timeframe:

1. A 1962 large scale wing cyclic test based on fleet experience and subsequent analysis established a safe life for the -135 force of 13,000 equivalent tanker hours. The results of this test are listed below:

   1962 Wing Cyclic Test Results

   o Test article failed catastrophically at

     - 10,800 spectra simulating 5.1 hour tanker missions

     - 55,080 simulated flight hours

   o 327 Fatigue cracks were discovered in the test article.

   o Established flight hours to 0.25 damage as

     - 55.080/4 = 13,770

     - Adjusted to 13,000 KC-135A flight hours including SEA usage

   o Four fatigue packages developed for fleet

By 1965 a safe life providing essentially a crack-free usage to 13,000 equivalent tanker hours was established based on test damage scatter factor of four and installation of four fatigue mod packages.

In 1968 a requirement for a full scale test was established to identify actions required to extend the life beyond 13,000 tanker flight hours. This test,known as the 1972 test,was more sophisticated and more representative of actual fleet usage. Due to unrepresentative high loads applied in the 1962 test which caused crack growth retardation, the 1972 test exhibited earlier and much more wide-spread cracking than the 1962 test. Review of the test and -135 fleet led to two conclusions:

1. Fatigue modifications would have to be expanded to assure safe life operation beyond 13,000 hours.

2. The wing lower surface would require replacement for life extension beyond 13,000 hours.

A summary of the 1972 test results are listed below.

1972 Cyclic Test Results

o Wing failed catastrophically (LWS 250) at

- 10,300 spectra (during the once per 100 spectra loading)
- 55,505 simulated flight hours

o A total of 8,276 fatigue crack locations were found in the wing lower surface.

o The number and significance of fatigue cracks found indicate a damage of 1.0 was reached prior to 40,000 flight hours of testing. Flight hours to 0.25 damage are less than 10,000 tanker flight hours.

Figure 37 gives a summary and general location of the wing lower surface fatigue failures found during the test and subsequent teardown inspection. The total number of cracks observed $\leq$ 0.01" and > 0.01" are summarized in Table 15 by structural detail.

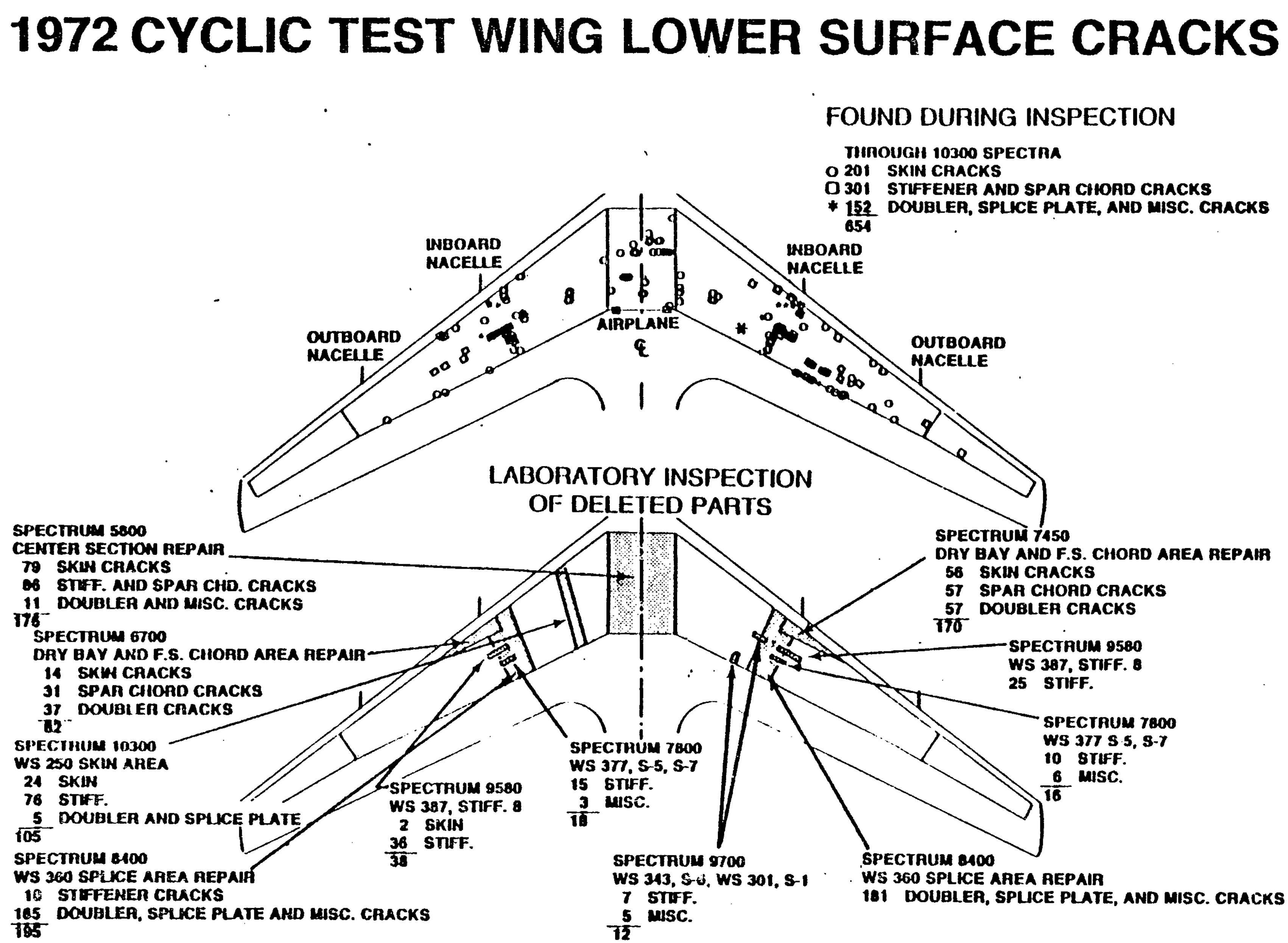


FIGURE 37 KC-135 Wing Lower Surface Fatigue Summary

TABLE 15 TOTAL WING LOWER SURFACE CRACKS;1972 CYCLIC TEST TEARDOWN.

STRUCTURAL DETAIL	AIRPLANE INSPECTION		DELETED * PARTS INSPECTION		TEARDOWN ** INSPECTION	TOTAL
	≤ .01"	> .01"	≤ .01"	> .01"		
SKIN	24	201	77	155	2062	2519
STIFFENER	79	238	66	231	2871	3485
SPAR CHORD		63	29	56	845	993
SPLICE PLATE AND DOUBLER	23	120	40	304	237	724
MISCELLANEOUS		32	7	38	478	555
TOTAL	126	654	219	784	6493	8276

* HOLE LOCATIONS IDENTIFIED DURING ON-AIRPLANE INSPECTIONS HAVE BEEN DELETED. LABORATORY METALLURGICAL INSPECTION RESULTS OF LWS 250 FRACTURE AT SPECTRUM 10300 ARE INCLUDED.

** LOWER WING SURFACE LWS 733 TO RWS 733

Since 1966, an upwards trend in the occurence of in-service fatigue cracks has been observed with approximately 30 instances of arrested unstable cracks, the longest of which was 43 inches. A summary of the significant fleet cracks are shown in Table 16.

To substantiate the fleet findings, lower surface parts from six aircraft which had reached 13,000 hours and had undergone lower skin replacement, were extensively inspected. Numerous cracks were discovered, mostly in the .05 to .10 inch range. A comparison of the fleet crack sizes with the teardown inspection results for the six 13,000 hours aircraft is shown in Table 17. Results of a teardown inspection for KC-135A, SN 57-1422 (shaded area) are also shown in Table 17. This aircraft had 11,500 hours at the time of inspection and is shown here to more accurately reflect the KC-135 fleet condition based on the 1975 decision to reskin the lower surface at 11,500 hours.

Based on the 1972 test results, fleet findings and an extensive evaluation conducted by a "blue ribbon" committee, it was concluded in 1973 that reskin at 11,500 hours would be required as insurance against fail safety degradation. Later findings in 1977 have again revised the lower wing reskin target at 8500 hours.

The reskin is being accomplished under ECP 405 modification. Figures 38 and 39 depict the various design changes and improvements associated with the ECP. A material change from the low damage tolerant 7178-T6 lower wing skin to the high damage tolerant 2024-T351 skin was a significant improvement in the ECP 405 mod. Elimination of splices and cutouts also added to the improvement.

Service experience is continually monitored and updated on the 707 and C/KC-135 fleet to identify durability problem areas that affect structural integrity. The known fleet problem areas relating to fatigue, stress corrosion, corrosion and miscellaneous items that have been identified from 707 and C/KC-135 fleet experiences are shown on Figure 40 and listed in Table 18. Figure 41 presents a cumulative plot of wing fatigue crack locations versus calendar year and Figure 43 shows the cumulative number of cracks at these locations versus calendar year.

TABLE 16 KC-135 SIGNIFICANT FLEET CRACKS

DESCRIPTION	SERIAL NO.	MODEL	CRACK LENGTH	FLIGHT HOURS	DATE FOUND	PRIOR REWORK
BOOST PUMP AREA						
SKIN CRACK, LWS 325, S-9 AND S-10	58-018	EC-135P	15.0	14,405	2-19-71	NO
SKIN CRACK LWS 325, S-9 AND S-10	58-065	KC-135Q	28.0	6,112	2-28-72	NO
SKIN CRACK, LWS 325, S-10	58-068	KC-135A	43.0	6,993	2-26-74	NO
DRY BAY AREA						
SKIN CRACK, RWS 392	64-14842	RC-135C	19.0	1,800	3-6-69	NO
SKIN CRACK, RWS 392	61-2665	WC-135B	19.0	11,076	2-3-70	REAMED AND COLD WORKED AT 5,384 HRS.
SKIN CRACK, RWS 392	61-2671	WC-135B	19.0	11,800	8-18-70	REAMED AND COLD WORKED AT 5,709 HRS.

TABLE 16 KC-135 SIGNIFICANT FLEET CRACKS(CONT'D)

DESCRIPTION	SERIAL NO.	MODEL	CRACK LENGTH	FLIGHT HOURS	DATE FOUND	PRIOR REWORK
DRY BAY AREA						
SKIN CRACK, RWS 392	61-2673	WC-135B	10.0	12,780	3-3-71	REAMED AND COLD WORKED AT 5,863 HRS.
SKIN CRACK, LWS 392	62-4129	VC-135B	4.25	12,993	3-1-72	REAMED AND COLD WORKED AT 6,020 HRS.
SKIN CRACK, RWS 392	64-14843	RC-135C	13.0	5,906	4-6-73	COLD WORKED PRODUCTION
SKIN CRACK, RWS 392	61-2663	RC-135S	19.0	14,785	3-12-76	REAMED AND COLD WORKED AT 12,628 HRS
T.E. RABBET CUT AREA						
SKIN CRACK, RWS 444	58-104	KC-135A	3.90	2,852	8-66	NO
SKIN CRACK, RWS 444	59-1505	KC-135A	3.90	2,818	4-67	NO
SKIN CRACK, LWS 444	58-090	KC-135A	3.90	3,554	6-67	NO
SKIN CRACK, RWS 444	59-1501	KC-135A	3.90	3,136	8-25-67	NO

TABLE 16 KC-135 SIGNIFICANT FLEET CRACKS (CONT'D)

DESCRIPTION	SERIAL NO.	MODEL	CRACK LENGTH	FLIGHT HOURS	DATE FOUND	PRIOR REWORK
T. E. RABBET CUT AREA						
SKIN CRACK, RWS 444	58-130 (LTF)	KC-135A	3.90	6,438	5-69	NO
SKIN CRACK, RWS 444	60-350	KC-135A	3.90	3,892	6-69	NO
SKIN CRACK, LWS 444	59-1482	KC-135A	3.90	3,987	7-69	NO
SKIN CRACK, LWS 444	59-1511	KC-135A	3.90	4,108	7-69	NO
SKIN CRACK, RWS 496	58-122	KC-135A	10.0	3,578	9-13-69	NO
SKIN CRACK, LWS 444	59-1494	KC-135A	3.90	5,697	7-24-73	T.O. 1C-135-752 AT 5,002 HOURS
SKIN CRACK, RWS 444	59-1477	KC-135A	3.90	7,091	5-4-76	T.O. 1C-135-752 AT 5,709 HOURS

TABLE 16 KC-135 SIGNIFICANT FLEET CRACKS (CONT'D)

DESCRIPTION	SERIAL NO.	MODEL	CRACK LENGTH	FLIGHT HOURS	DATE FOUND	PRIOR REWORK
STIFFENER TO RIB ATTACHMENT AREA						
STIFFENER CRACK, LWS 558, S-8	59-1507	KC-135A	3.0	4,777	11-22-72	NO
STIFFENER CRACK, RWS 558, S-3	59-1470	KC-135Q	COMPLETE	6,208	11-16-72	NO
CENTER SECTION AREA						
SKIN CRACK, LH BBL 65, S-4	58-113	KC-135A	7.4	8,061	5-11-76	NO
STIFFENER RUNOUT AREA						
SKIN CRACK, LWS 570, S-12	55-3141	KC-135A	14.0	7,295	4-8-74	NO
SKIN CRACK, WS 452.3, S-15	64-14848	RC-135U	7.50	7,096	12-16-75	NO
MISCELLANEOUS						
SKIN CRACK, WS 250, S-8	58-020	KC-135A	29.0	4,720	3-2-72	PRIOR -3 FLUSH REPAIR INSTALLED

TABLE 17 KC-135 COMPARISON OF TEARDOWN CRACK SIZES TO FLEET CRACK SIZES

AIRPLANE TMS	AIRPLANE SN	CRACK LENGTH ~ INCHES <.05	.05 TO .10	.11 TO .25	.26 TO .50	.51 TO 1.00	1.01 TO 2.00	2.01 TO 3.00	3.01 TO 5.00	5.01 TO 10.00	OVER 10.01	TOTAL
RC-135D	60-356	20	16	9	4	6	0	0	0	0	0	55
RC-135D	60-362	470	64	16	3	4	0	0	1	0	0	558
KC-135A	55-3137	[1] 250	10	5	4	1	0	0	0	0	0	270
KC-135A	59-1481	419	21	4	1	0	0	0	0	0	0	445
WC-135B	61-2672	4	9	5	0	1	0	0	0	0	0	[2] 19
WC-135B	61-2667	350	58	3	0	0	0	0	0	0	0	411
KC-135A	57-1422	[3] 672	22	1	2	1	0	0	0	0	0	698
FLEET		41	20	10	11	0	4	7	11	5	10	119

[1] OCAMA REPORTED 209 OF THE 250 AS STRESS CRACKS COMPARABLE TO MACRO CRACKS AND CIRCUMFERENTIAL CRACKS FOUND BY BOEING DURING MET LAB INSPECTIONS.

[2] AFML REPORTED A TOTAL OF 594 CRACKS OF WHICH 19, SHOWN HERE, WERE CONFIRMED AS FATIGUE; 82 WERE CLASSED AS POSSIBLE FATIGUE AND 498 WERE RADIAL, BORE OR CIRCUMFERENTIAL CRACKS.

[3] ONLY 11 CRACKS < .01

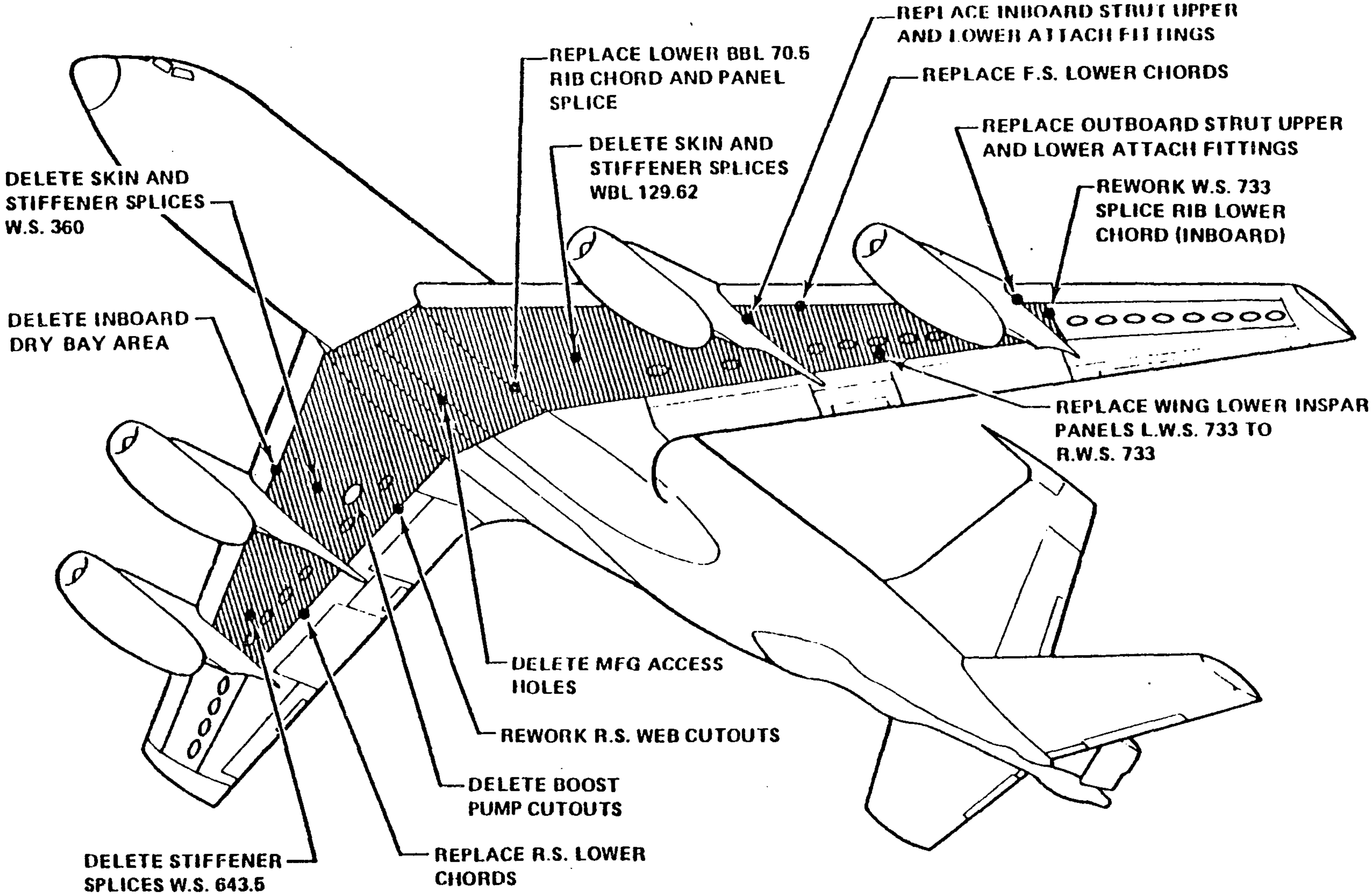


FIGURE 38 C/KC-135, ECP 405 Modification

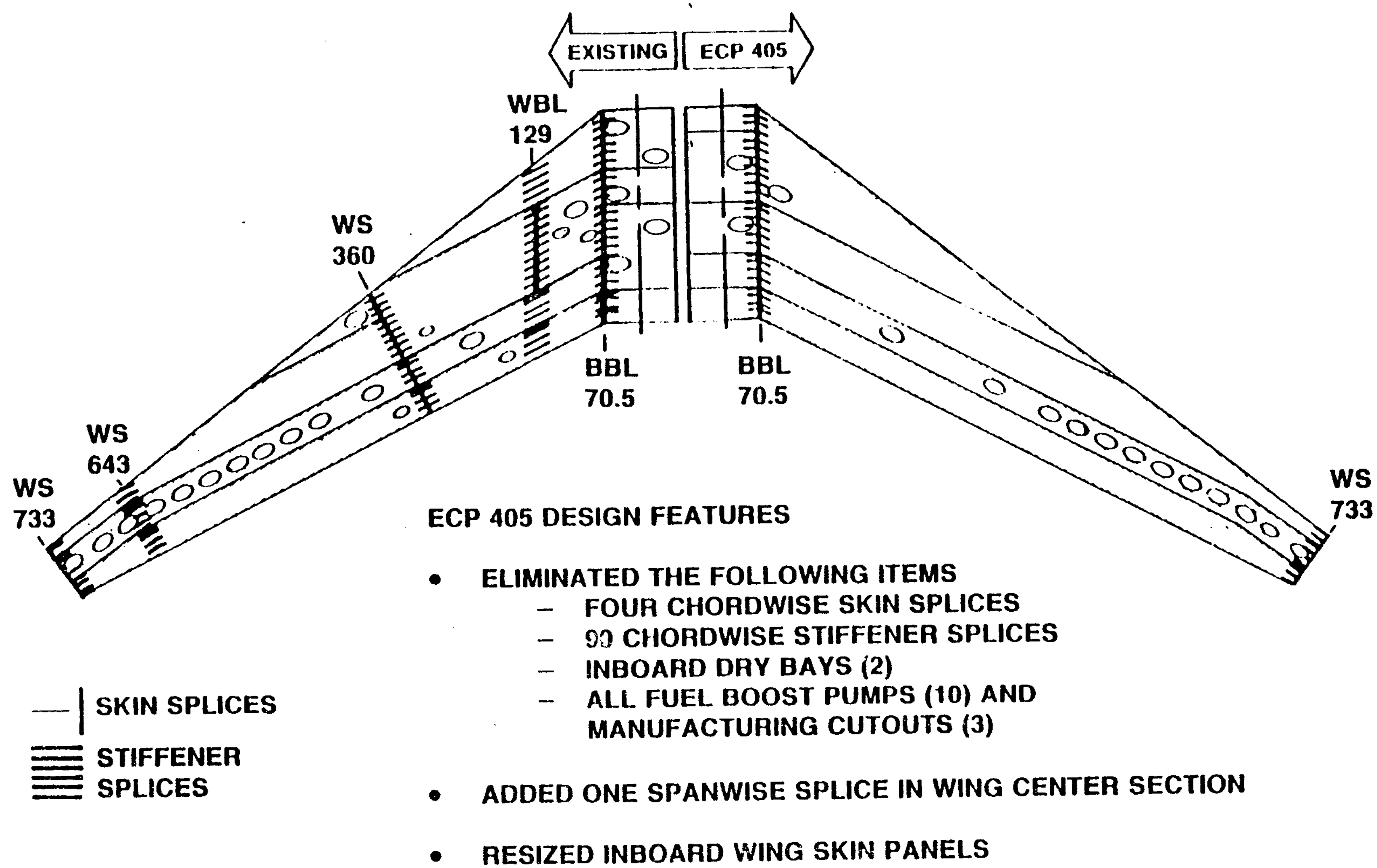


FIGURE 39 C/KC-135, ECP 405 Configuration Comparison

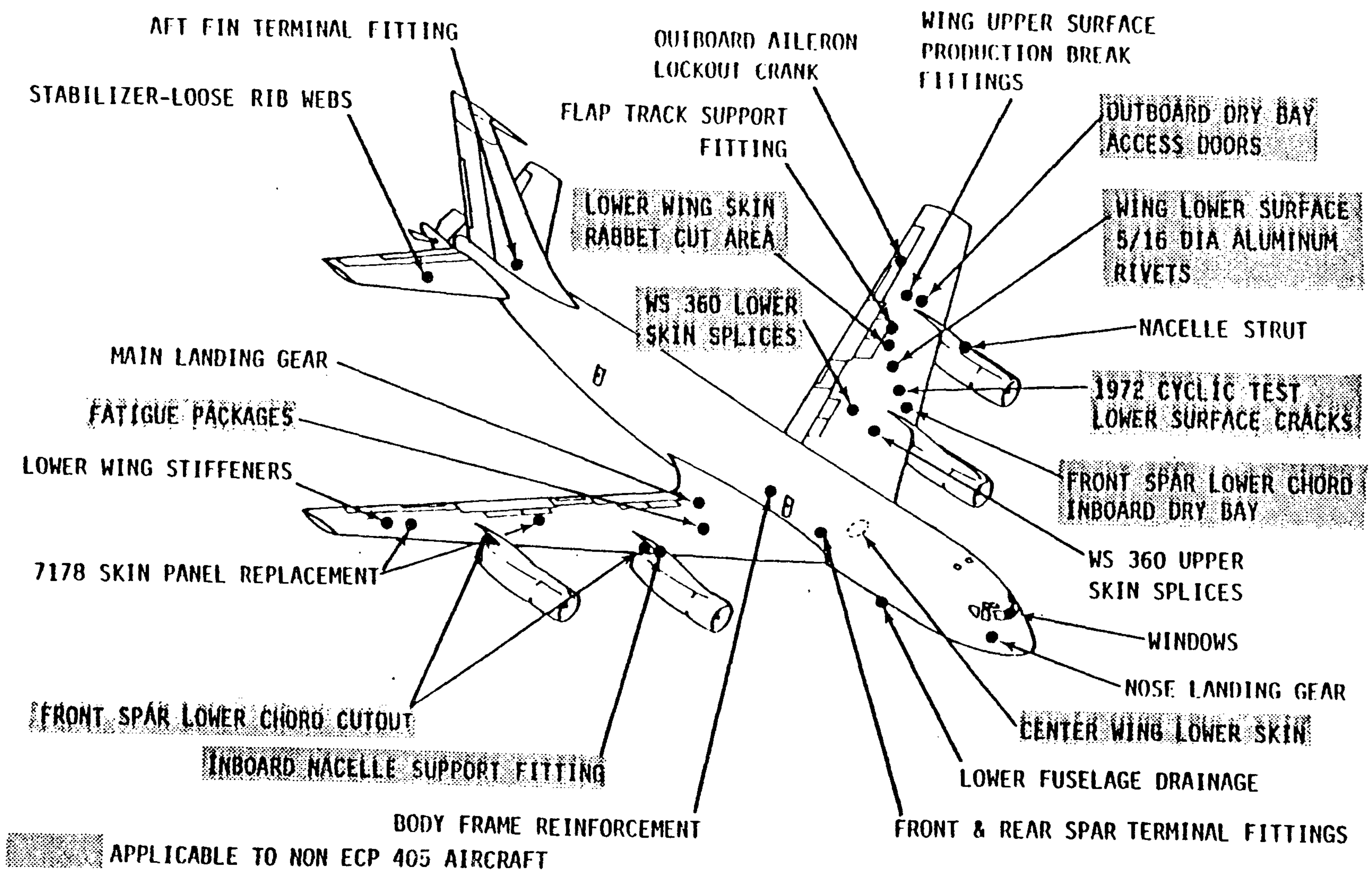


FIGURE 40 C/KC-135 Problem Areas

TABLE 18 KC-135 KNOWN STRUCTURAL PROBLEM AREAS

- FATIGUE

  - ECP 449 LOWER WS 360 EXTERIOR SPLICE PLATES (REF. S.B. 2592)
  - ECP 434 FATIGUE PACKAGE I FOR ECP 330-10 AIRPLANES
  - ECP 439 FATIGUE PACKAGES II, III & IV FOR ECP 330-10 AIRPLANES
  - WS 360 UPPER SURFACE SKIN SPLICE ▷ (REF. S.B. 3160)
  - FS LOWER CHORD CUTOUT INSPECTION (REF. S.B. 3136)
  - FS LOWER CHORD INBOARD DRY BAY AREA (REF. S.B. 2765)
  - LOWER WING SKIN RABBET CUT AREA
  - LOWER WING CENTER SECTION SKIN (REF. S.B. 3230)
  - FLAP TRACK SUPPORT FITTINGS (REF. S.B. 3222)
  - WING AFT TERMINAL FITTING LOWER FLANGE HOLES
  - OUTBOARD AILERON LOCKOUT CRANK
  - 1972 CYCLIC TEST WING LOWER SURFACE CRACKS
  - ECP 431 FATIGUE PACKAGE - BODY FRAME REINFORCEMENT (STA. 640-800)
  - INSPECT AFT FIN TERMINAL FITTING ATTACH HOLES ▷ (REF. S.B. 3216)
  - STABILIZER LOOSE RIB WEBS
  - MAIN LANDING GEAR UPLOCK SUPPORT STRUCTURE (REF. S.B. 2411)
  - ECP 448 MAIN LANDING GEAR TRUNNION SUPPORT FITTING REWORK (REF. S.B. 3056)
  - NOSE GEAR BRACKET REPLACEMENT (REF. S.B. 3207)
  - INBOARD NACELLE FRONT SPAR SUPPORT FITTING
  - NACELLE STRUT (REF. S.B. 3183 & 2958)

▷ POTENTIAL PROBLEM AREA

TABLE 18 KC-135 KNOWN STRUCTURAL PROBLEM AREAS (CONT'D)

- STRESS CORROSION
  - ECP 438 FRONT AND REAR SPAR TERMINAL FITTINGS - BUSHING INSTALLATION
  - REAR SPAR TERMINAL FITTING REPLACEMENT ECP 405 AIRCRAFT
  - WING LOWER STIFFENERS
  - MAIN LANDING GEAR LOCK SUPPORT FITTING (REF. S.B. 2837)
  - MAIN LANDING GEAR SIDE BRACE (REF. S.B. 2638)
  - NOSE GEAR OUTER CYLINDER (REF. S.B. 2005)
- CORROSION
  - WING UPPER SURFACE PRODUCTION BREAK FITTINGS
  - DRAINAGE OF LOWER FUSELAGE (REF. S.B. 3172)
  - MAIN LANDING GEAR TRUNNION BEAM
  - MAIN LANDING GEAR TRUCK BEAM
  - MAIN LANDING GEAR AXLES
  - MAIN & NOSE LANDING GEAR STRUTS
- MISCELLANEOUS
  - NEW SPARE WINDOWS
  - AVAILABILITY OF 7178 PLATE FOR WING SKIN PANELS REPLACEMENT
  - OUTBOARD DRY BAY ACCESS DOORS
  - MAIN LANDING GEAR BUSHING WEAR AND REPLACEMENT
  - WING LOWER SURFACE - 5/16" DIA. ALUMINUM RIVETS

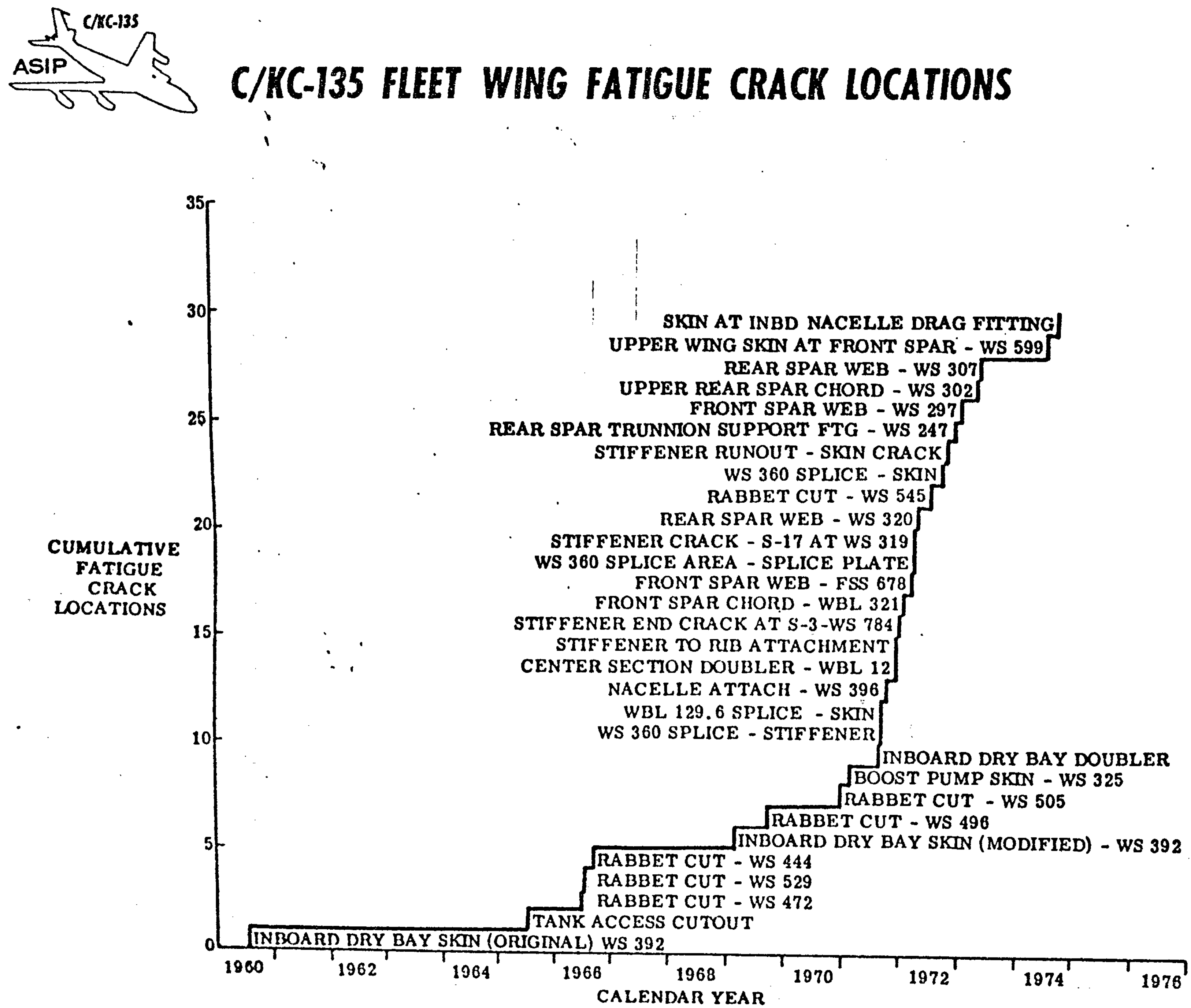


FIGURE 41 C/KC-135 Fleet Wing Fatigue Crack Locations

C/KC-135 FLEET WING FATIGUE CRACKS

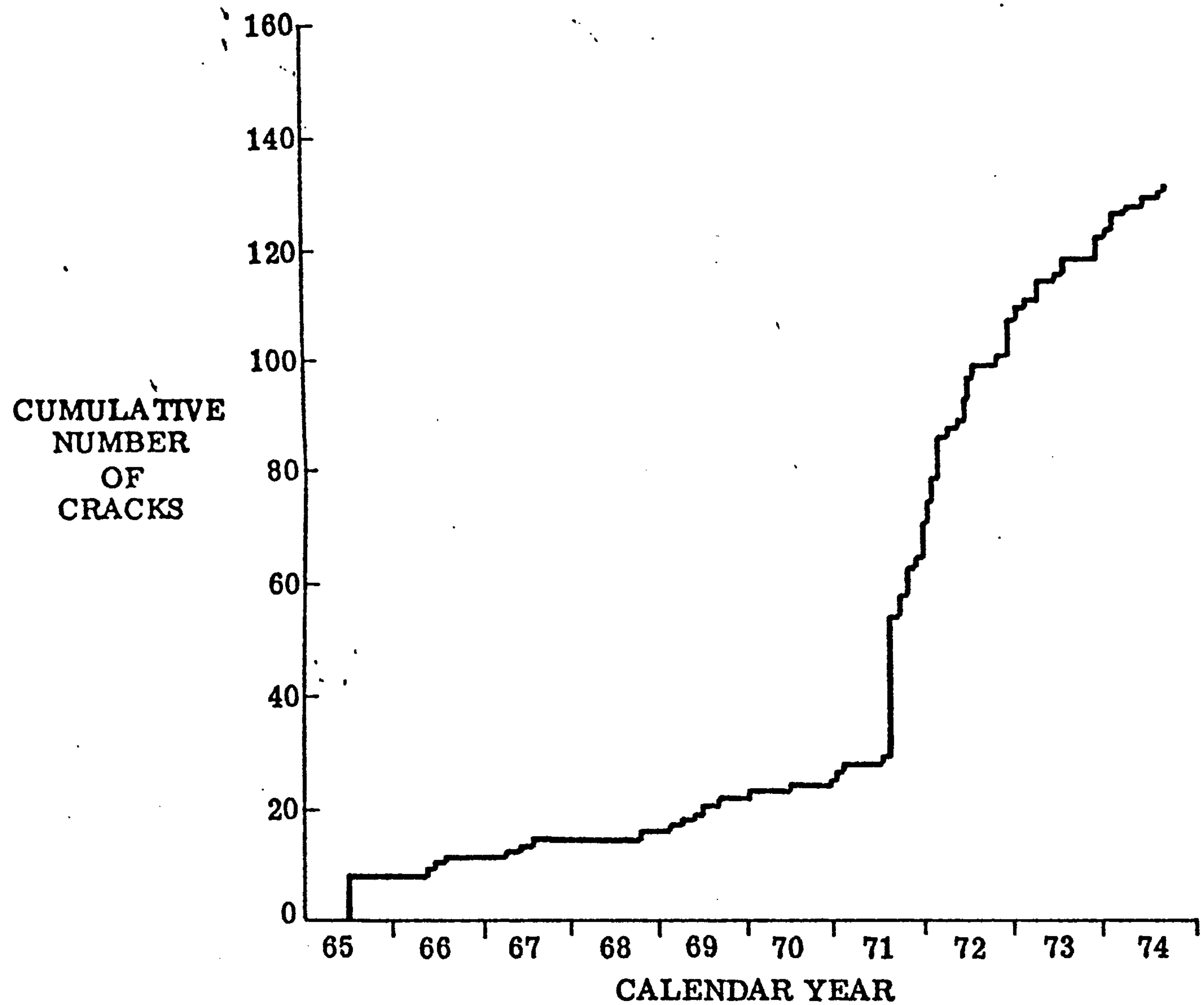


FIGURE 42 C/KC-135 Fleet Wing Fatigue Cracks Vs Calendar Year

The KC-135 Cyclic Test Article teardown inspection results for the wing are detailed in Reference (9). A **summary extracted** from that document is presented below and in Tables 19 and 20.

1. From Table 19, it can be seen that of 457 wing parts inspected, 203 or 44 percent exhibited some form of cracking at a total of 639 locations. Twenty-nine percent of the locations had some crack dimension in a hole bore or countersink surface. The balance/majority of the wing structure cracks occurred on skin and substructure surfaces adjacent to fastener holes.

2. From Table 20 , it can be seen that of 772 fuselage and empennage parts tested, 75 or approximately 10 percent exhibited some form of cracking at a total of 255 locations. Eighty-eight percent of the locations had some crack dimension in a hole bore or countersink surface. The balance/minority (12%) of the body/empennage cracks occurred on skin and substructure surfaces adjacent to fastener holes.

3. Four crack locations were noted in the empennage section. No cracks were found in any of the steel parts inspected at AFML with the flourescent magnetic particle method.

4. The wing section exhibiting the greatest percentage of cracked parts was W34, just outboard of Wing Station 558, at the rear of the wing. However, W21 exhibited, by far, the greatest number of crack indications (WS360) totaling 186.

5. The body/empennage section exhibiting the greatest percentage of cracked parts was B48, located at Body Station 820 slightly aft of the wing. Section B41, however, had 63% more crack indications than B48, with a total of 31. This section is located near the base of the wing on the side of the fuselage above the wing.

6. A final determination of the degree of significance of these data will await the results of combining the data of all the participants so as to obtain a complete picture of the condition of the aircraft. The AFML data, however, indicate that by far, the majority of the flaws found had not propagated beyond approximately 100 to 200 **mils** in length.

TABLE 19 SUMMARY OF FATIGUE DAMAGE FOUND IN WING SECTIONS

	Section No.	Parts Inspected	Parts Cracked	% Parts Cracked	Crack Locations	In Hole Crack Locations *	Non Hole Crack Locations **	% in Hole Crack Locations
	W6	6	2	33	5	5	0	100
	W9	42	20	48	57	7	50	12
	W12	23	2	9	4	4	0	100
	W15	23	13	57	54	9	45	17
	W18	16	10	63	26	13	13	50
	W21	66	44	67	186	70	116	38
	W22	49	34	69	104	19	85	18
	W24	31	7	23	21	5	16	24
	W27	19	13	68	21	10	11	48
	W31	7	1	14	2	0	2	0
	W34	11	9	82	27	6	21	22
	W35	11	5	45	7	0	7	0
	W37	17	5	29	16	4	12	25
	W40	10	0	0	0	0	0	0
	W42	62	22	35	54	11	43	20
	W44	7	0	0	0	0	0	0
	W46	15	3	20	13	9	4	69
	W55	21	2	10	2	1	1	50
	W58	16	11	69	30	7	23	23
	W61	5	0	0	0	0	0	0
Total	20	457	203	44% Avg	629	180	449	31% Avg

* Crack locations with some crack dimension in a fastener hole bore or countersink surface.

** Crack locations with no crack dimension in a fastener hole bore or countersink surface.

TABLE 20 SUMMARY OF FATIGUE DAMAGE FOUND IN FUSELAGE AND EMPENNAGE SECTIONS

	Section No.	Parts Inspected	Parts Cracked	%Parts Cracked	Crack Locations	In Hole Crack Locations	Non Hole Crack Locations	% in Hole Crack Locations
	B2	20	2	10	4	4	0	100
	B5	27	2	7	13	12	1	92
	B8	30	4	13	6	6	0	100
	B11	26	3	12	19	7	12	37
	B14	21	2	10	2	1	1	50
	B17	24	4	17	13	11	2	85
	B20	31	5	16	18	13	5	72
	B23	36	2	6	9	9	0	100
	B26	26	3	12	18	18	0	100
	B29	24	2	8	10	10	0	100
	B32	34	7	21	23	21	2	91
	B35	15	1	7	1	1	0	100
	B38	17	0	0	0	0	0	0
	B41	27	2	7	31	31	0	100
	B44	22	1	5	1	1	0	100
	B48	41	9	22	19	16	3	84
	B50	15	0	0	0	0	0	0
	B53	31	3	10	7	7	0	100
	B56	18	0	0	0	0	0	0
	B59	27	7	26	15	15	0	100
	B63	29	6	21	16	15	1	94
	B65	35	2	6	3	3	0	100
	B68	7	0	0	0	0	0	0
	B71	27	2	7	15	15	0	100
	B74	63	0	0	0	0	0	0
	B77	26	0	0	0	0	0	0
	B80	20	1	5	1	0	1	0
	B83	19	1	5	4	4	0	100
	E3	18	1	6	1	0	1	0
	E6	16	3	19	6	5	1	83
Total	30	772	75	10% Avg	255	225	30	62% Avg

7. The use of no less than 20X magnification for inspection was definitely required despite the clean etched part surfaces. This magnification significantly limited depth of field for viewing purposes, but there was no other possible alternative.

3.4.4 C-5A Aircraft

The C-5A aircraft (Ref. 10) is a high-altitude, long-range, high-subsonic-speed, heavy-logistic transport air vehicle capable of worldwide, all weather operation.

The service life design goal for the C-5A was 30,000 flight hours, 12,000 landings, and a total of 5950 pressure cycles all representing 15 mission profiles.

As of 1974, there were five major fatigue test articles designed to assess the fatigue service life of the C-5A. The original goal for these specimens was to test to four lifetimes which would equate to 120,000 cyclic test hours. The five major tests were as follows:

1. Full-scale Fuselage and Wing Assembly. Cyclic testing on the wing of this test article was terminated at 24,000 cyclic test hours due to general fatigue cracking.

2. Full-scale Aft Fuselage-Empennage (FS 1964). This test article completed 60,000 cyclic test hours.

3. Main Landing Gear - The test article also included the fuselage support structure. This test article completed four lifetimes (48,000 landings) of testing.

4. Nose Landing Gear - The test article also included fuselage support structure. Only portions of the test article completed four lifetimes (48,000 landings) due to major failures caused by fatigue and stress corrosion.

5. Expedited Wing - This test article was programmed to provide an accelerated test of the wing structure. Cyclic testing was terminated after 60,000 cyclic test hours due to a general cracking condition.

Based on these test program results, approximately 50 control points have been selected to identify the critical areas where cracking either occurred in test or might be expected to occur in service. These areas are shown in Figures 43, 44 and 45. Figure 43, for the wing surface, define the type structure where cracking has or is expected to occur.

Extensive cracking in the C-5A wing and projection that the existing wing would not meet the required service life resulted in an "H" model redesign wing. The new wing is being designed to meet the current durability and damage tolerance requirements. Significant material changes as well as design improvements have been made to provide a much improved durability and damage tolerant wing. Material comparisons between the old and new design wings are as follows:

Type Structure	Material Selection	
	Old Design	New Design
o Panels	7075-T651 Extr.	7175-T7351 Extr.
o Beam Caps	7075-T651 Extr.	7175-T7351 Extr.
o Webs	7075-T6	7475-T651
o Rib Caps	7075-T6 Extr.	7075-T73
o Forgings	7075-T6	7075-T73 7175-T73

An extensive individual aircraft service life monitoring program (IASLMP) is being used to monitor the service life of the C-5A aircraft. Table 21 presents a summary of fleet service cracks as related to IASLMP data.

Analytical condition inspections (ACI) have also been used to monitor the durability of the C-5A structure. These data for Calendar Years 1976 and 1977 are included in Figures 46 through 50.

WING UPPER SURFACE CONTROL POINTS

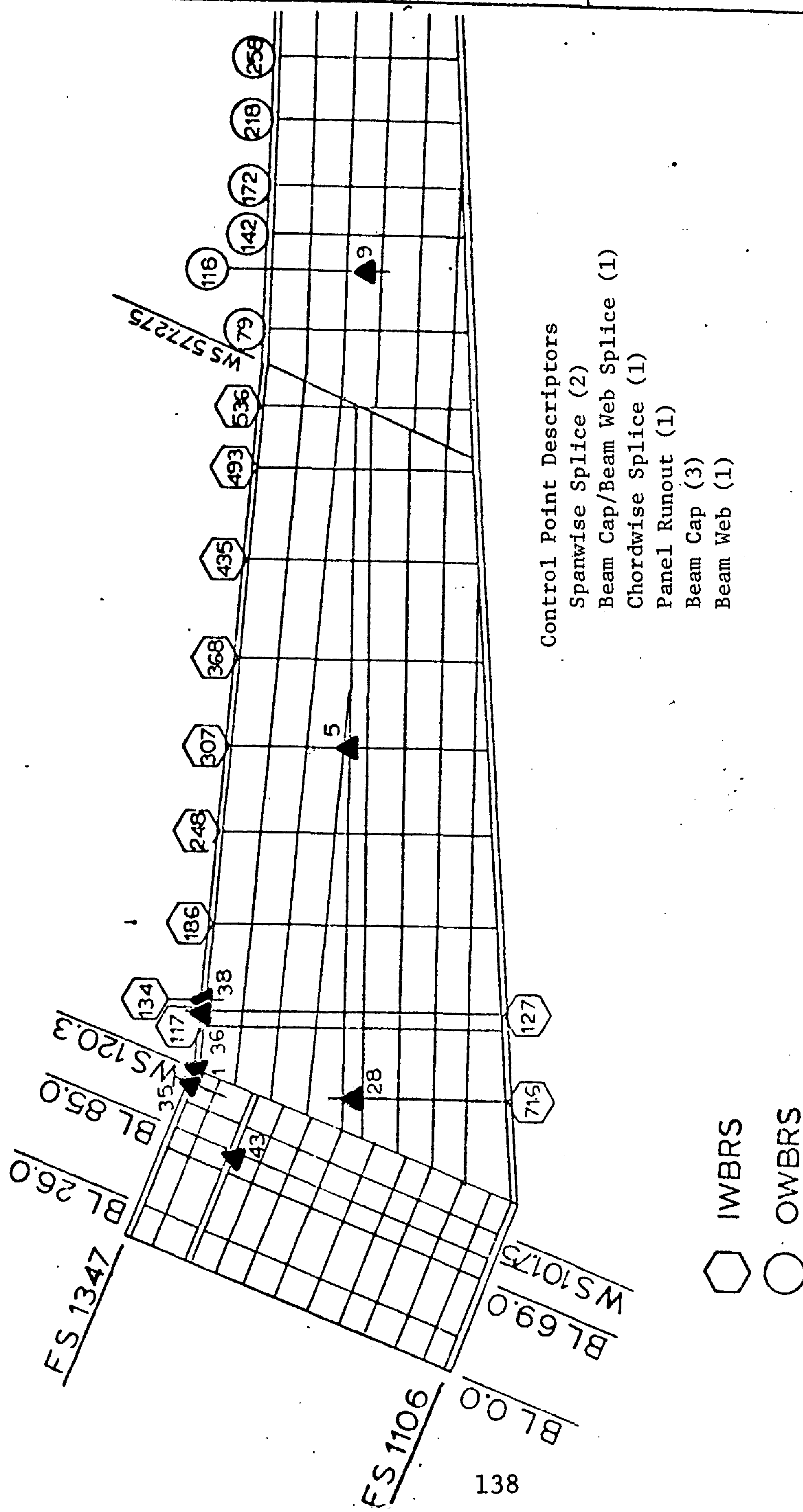


FIGURE 43 C-5A Wing Upper Surface Control Points

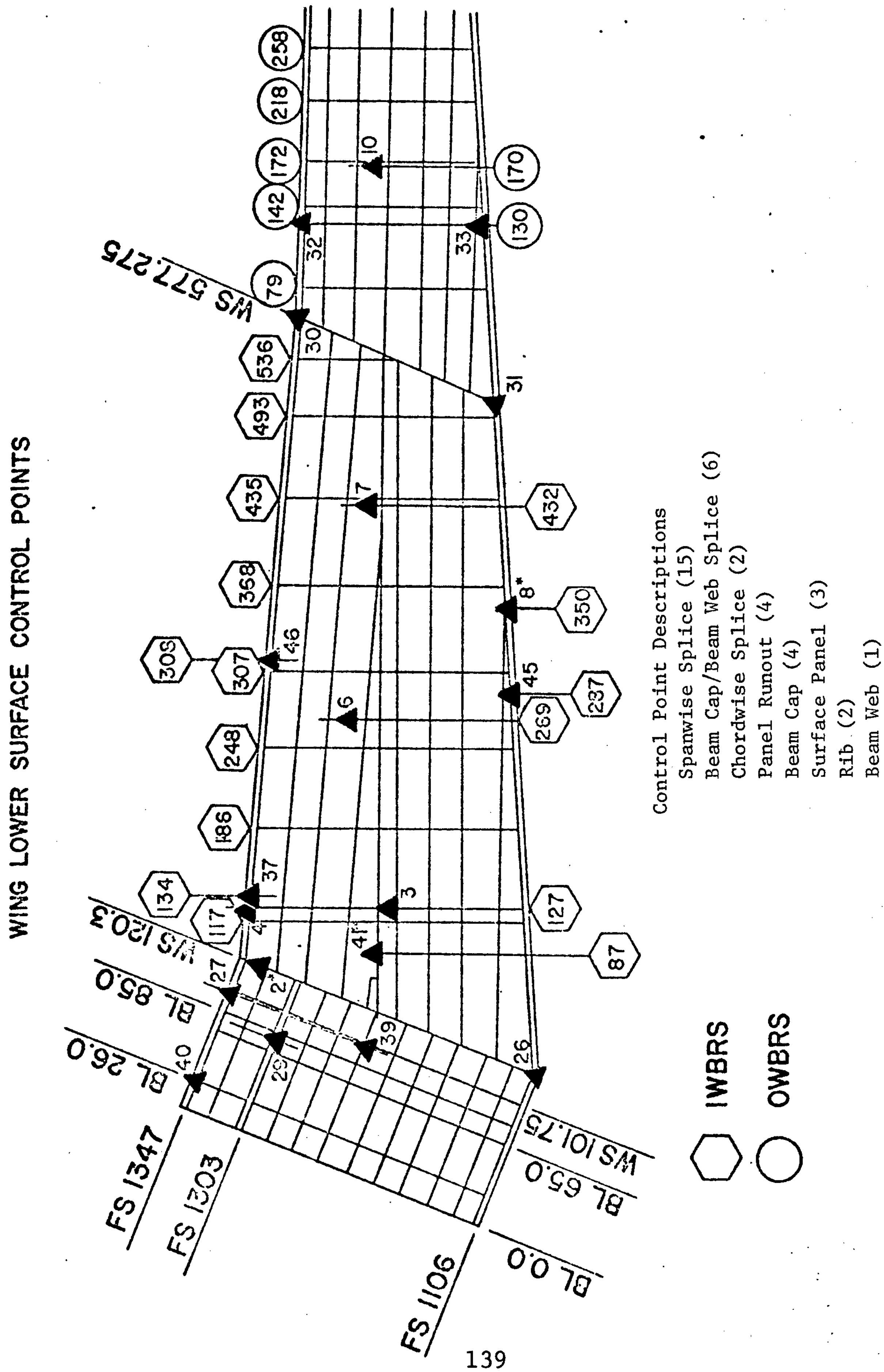


FIGURE 44 C-5A Wing Lower Surface Control Points

FIGURE 3.3 AIRCRAFT STRUCTURAL CONTROL POINTS (EXCLUDING WING)

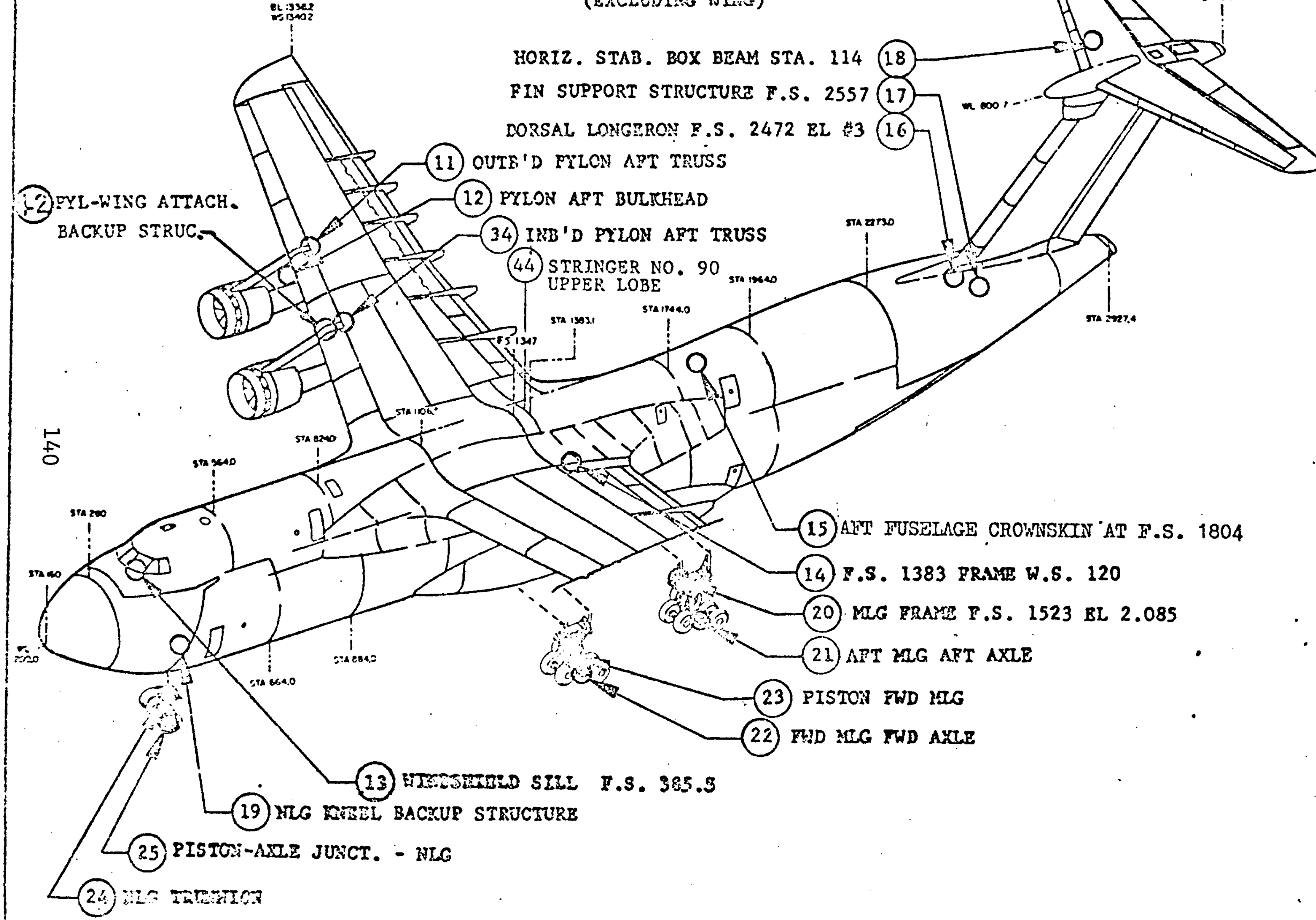
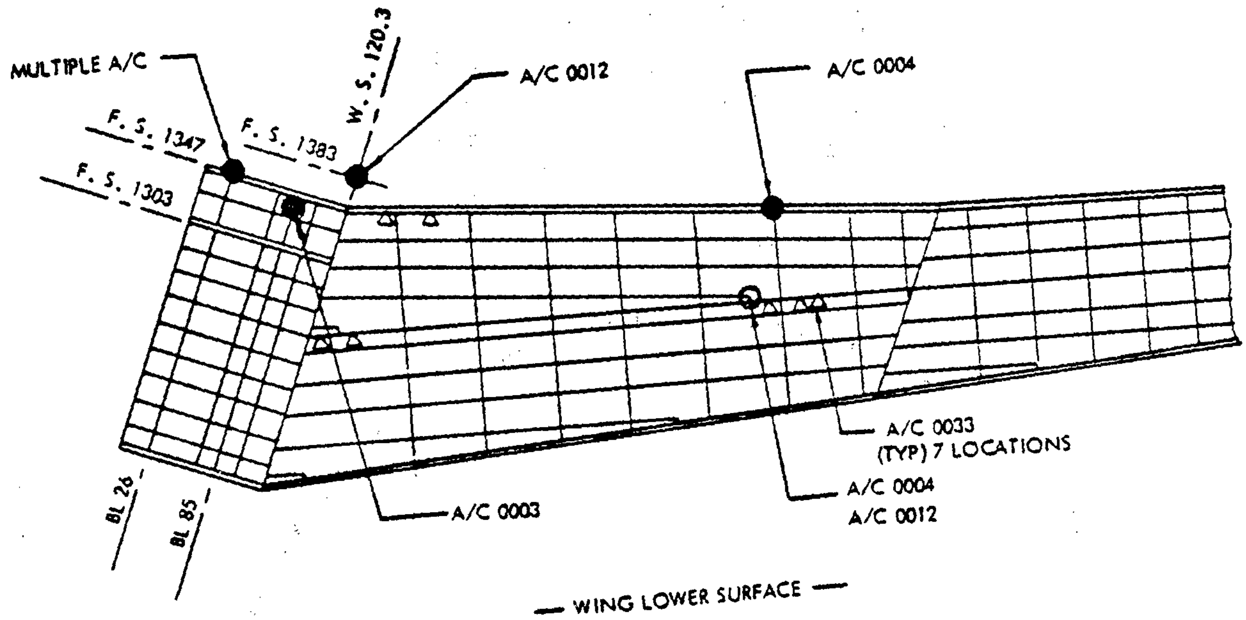


FIGURE 45 C-5A Aircraft Structural Control Points (Excluding Wing)

TABLE 21 C-5A SUMMARY OF FLEET SERVICE CRACKING

LOCATION OF REPORTED CRACKS IN SERVICE AIRCRAFT

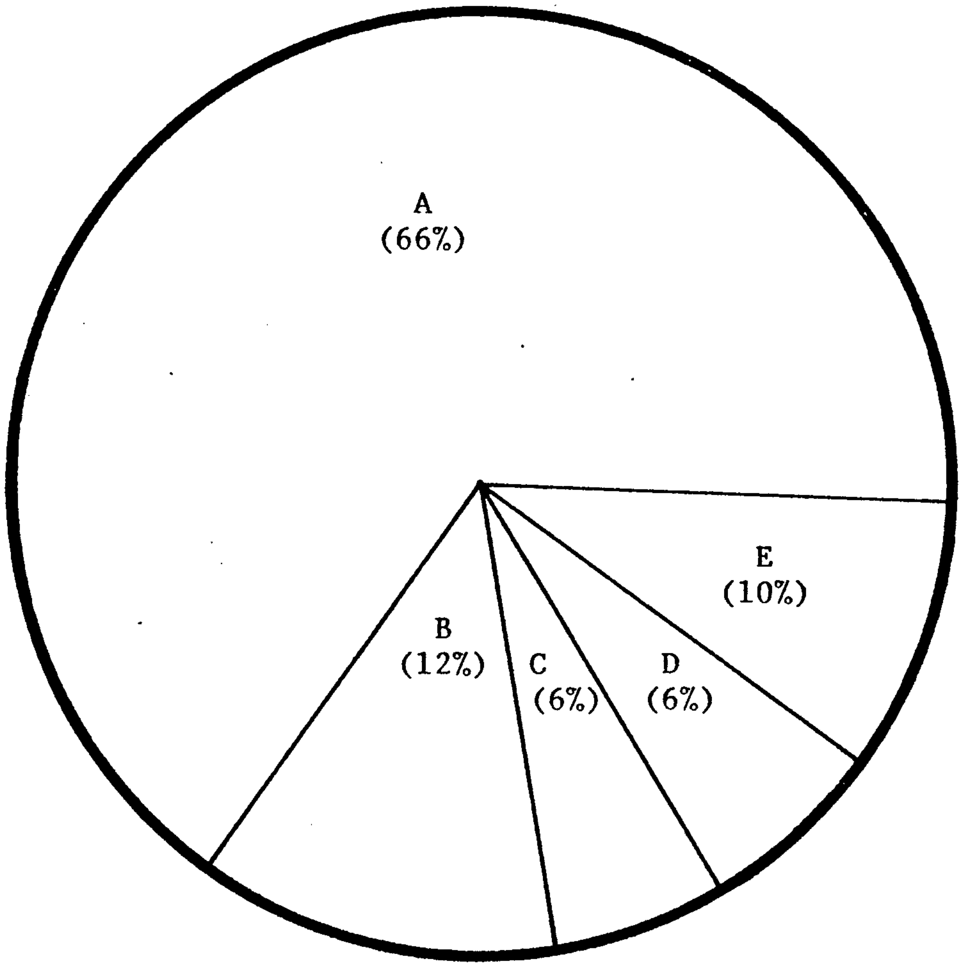


— WING LOWER SURFACE —

DESCRIPTION OF SERVICE AIRCRAFT CRACKS

LAC NO.	PROBLEM DESCRIPTION	DATE	FLIGHT HOURS	IASLMP DAMAGE	TCTO/ REMARKS
40 A/C	CTR WING REAR BEAM WEB BL 26 RADIUS	FEB 1972 & ON	543 TO 2200	.077 (MEAN)	1C-5A-1635 OR DIAMOND DOUBLER REPAIR
0012	INNER WING LOWER SURFACE PANEL #4 RUNOUT	OCT 1972	1798	.076	1C-5A-1295 MEASURED DEFECT (SUSPECT CRACK)
0012	F.S. 1383 MAIN FRAME FAILSAFE STRAP CRACKED (L & R)	OCT 1972	1798	.316	1C-5A-1289
0003	CTR WING LOWER SURFACE PRESSURE KICK FITTING	MAY 1972	883	.124	1C-5A-1393 DISCREPANT PART
0004	INNER WING LOWER REAR SPAR CAP AT TRAILING EDGE ATTACHMENT	JAN 1974	1696	.183	1C-5A-1296
0033	INNER WING SPANWISE SPLICE & WEB TO RISER FASTENER PULLING INSPECTION	OCT 1975	4158	.156	1C-5A-1786 7 NDI INDI-CATIONS
0004	INNER WING LOWER SURFACE PANEL #4 RUNOUT	MAR. 1978	3090	.120	1C-5A-1855 CONFIRMED. FATIGUE CRACKS

C-5A TRANSPORT
SUMMARY OF ANALYTICAL CONDITION INSPECTION (ACI) RESULTS
FOR CY 1976



DETAIL-MAJOR INCIDENTS

A 32 Incidents Cracked Step Support Fittings

B 6 Incidents Cracked Handrail Support Fittings

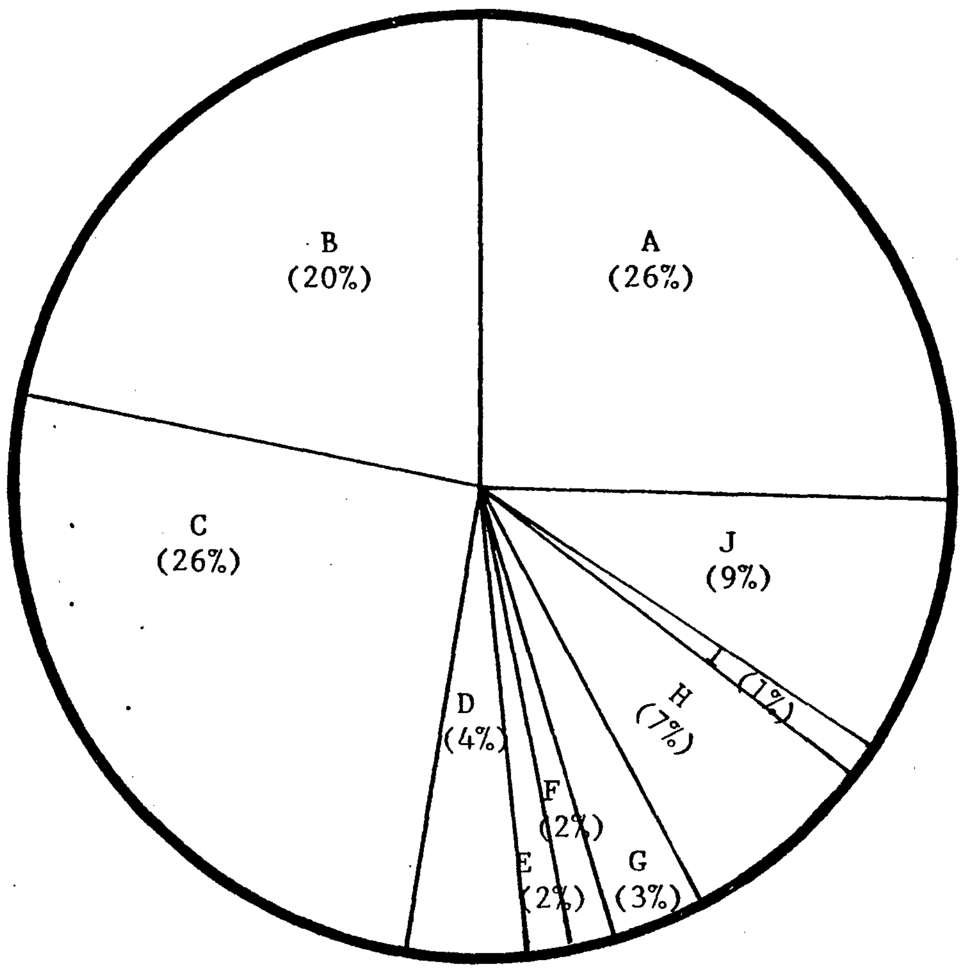
C 3 Incidents Other Cracking

D 3 Incidents Face Sheet Cracks in Galley

E 5 Incidents Floor Panel Delaminations

FIGURE 46 C-5A ACI Summary For CY 1976 (Major Incidents)

C-5A TRANSPORT
ANALYTICAL CONDITION INSPECTION RESULTS FOR CY 1976



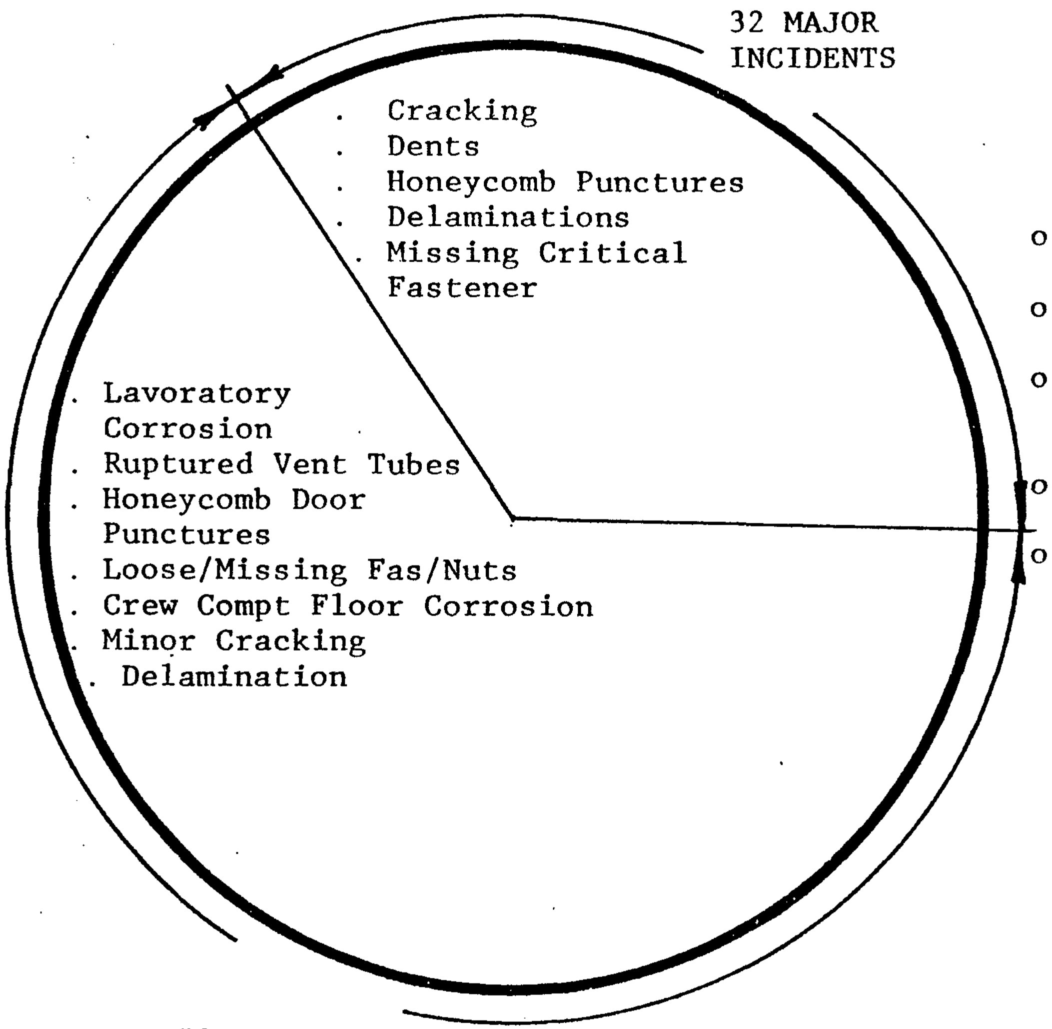
121 MINOR INCIDENTS
(FROM INSPECTION OF 10 A/C)

DETAIL MINOR INCIDENTS

A	31 Incidents of Excessive Wear on Cargo Door Latches
B	26 Incidents Broken/Missing Fas.
C	31 Incidents Honeycomb Punctures
D	5 Incidents Improper Honeycomb Repair
E	2 Incidents Water Entrapment'
F	2 Incidents Minor Corrosion
G	4 Incidents Other Cracking
H	8 Incidents Loose/Torn Pressure Seals
I	1 Incident Deep Nick in Fastener
J	11 Incidents Cracked Support Braces, Brackets & Clips

FIGURE 47 C-5A ACI Summary For CY 1976 (Minor Incidents)

C-5A TRANSPORT
SUMMARY OF ANALYTICAL CONDITION INSPECTION (ACI) RESULTS
FOR FY 1977

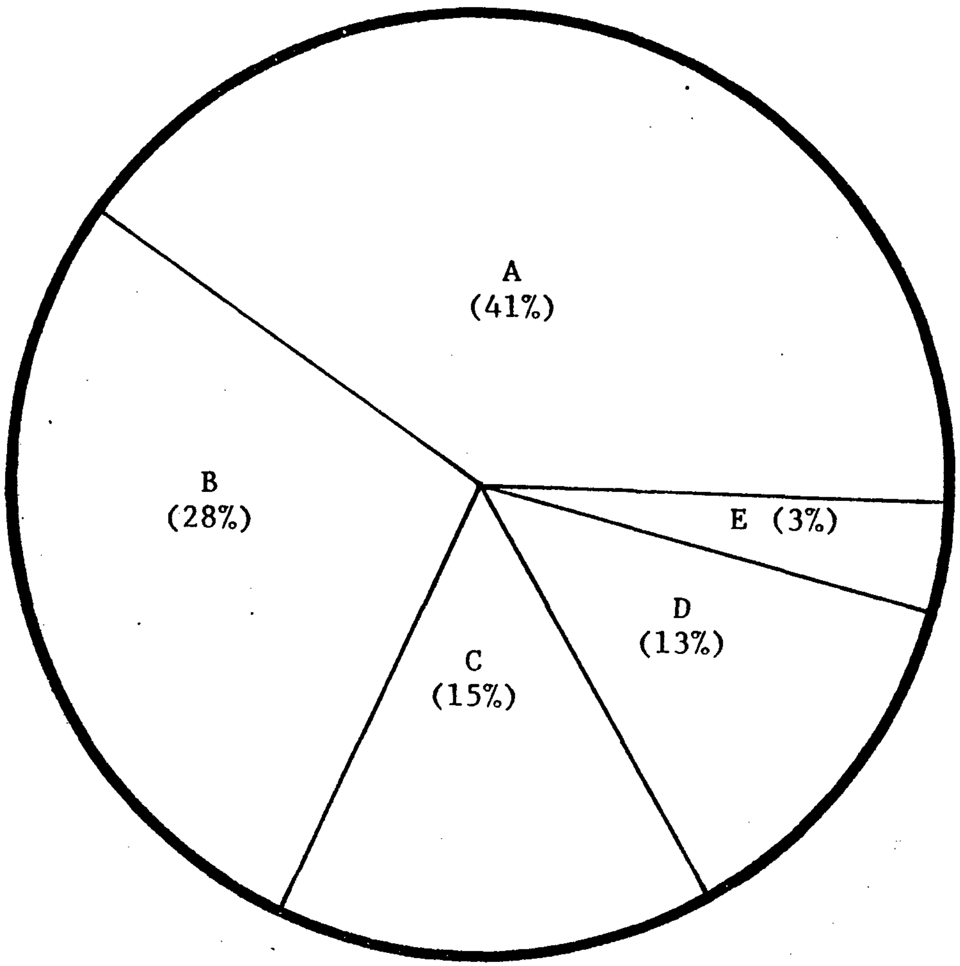


COMMENTS

- o 6 Aircraft
- o Normal Fleet Usage
- o Flying Hours 3147 - 3699
- o Average - 3395
- o No Critical Incidents

FIGURE 48 C-5A ACI Summary For Fy 1977

C-5A TRANSPORT
ANALYTICAL CONDITION INSPECTION RESULTS FOR FY 1977



32 MAJOR INCIDENTS
(FROM INSPECTIONS OF 6 A/C)

DETAIL - MAJOR INCIDENTS

A 13 Incidents of Honeycomb Floor Panel Cracks, Dents & Delaminations

B 9 Incidents of Flight Engineers Floor Cracks, Dents, Punctures & Delaminations

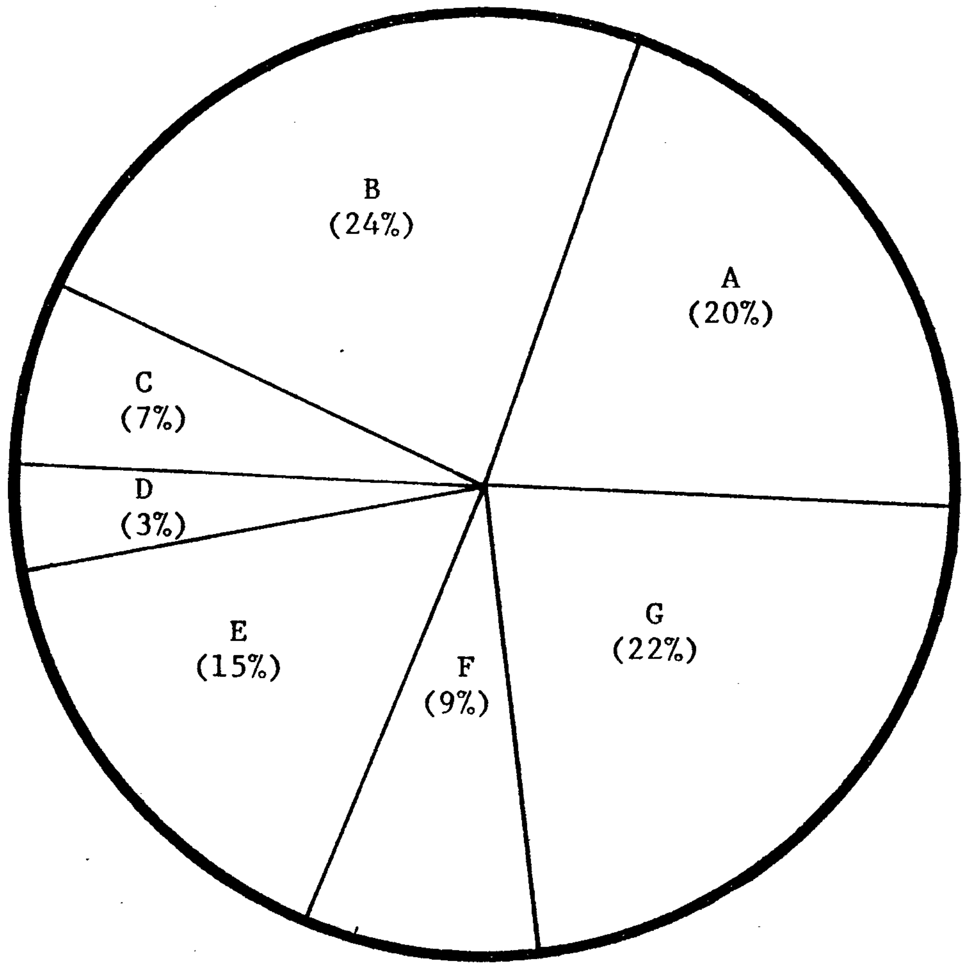
C 5 Incidents of Ramp Interior, Cracking and Missing Fastener

D 4 Incidents of Water Entrapment in Spoiler Honeycomb Panels Which Eventually Leads to Bond Delamination.

E 1 Other

FIGURE 49 C-5A ACI Summary For FY 1977 (Major Incidents)

C-5A TRANSPORT
ANALYTICAL CONDITION INSPECTION RESULTS FOR FY1977



DETAIL - MINOR INCIDENTS

- A 12 Incidents of Corrosion,
- B 14 Incidents Loose, Missing, And/Or Damage Fasteners
- C 4 Incidents Ruptured Lines
- D 2 Incidents of Punctured Honeycomb Panels/Doors.
- E 9 Incidents of Elongated Holes
- F 5 Incidents of Cracks/Scratches
- G 13 Other

59 MINOR INCIDENTS
(FROM INSPECTION OF 6 A/C)

FIGURE 50 C-5A ACI Summary For FY 1977 (Minor Incidents)

3.5 ATTACK AIRCRAFT

3.5.1 A-10 Aircraft

The A-10 aircraft (Ref. SA160R9401, Ref. 11) is a subsonic single place airplane with a primary mission of providing sustained close air support.

The design service life for the A-10 was 6000 hours with a scatter factor of 2.0.

The damage tolerance criteriawere established prior to the MIL-A-83444 specification and consisted of two parts.

1. Get home residual strength capability following battle damage from specified threats.

2. Safe life in the presence of manufacturing defects or service induced flaws.

During the two lifetimes of fatigue testing a number of premature failures occurred. A discussion of these results is presented below.

At .819 lifetimes, the following discrepancies were noted:

- Fuselage frame 405 cracking
- Hole cracking in lower closure member
- Pulled through door fasteners at Frame 405
- Bend radius and rivet hole cracking in angles at Frame 405
- Numerous fastener failures attaching skin in area of Frame 405
- Cracked panels on side of inner fuel tank floor at Station 405

A thorough investigation revealed that the following factors were the major cause of this failure.

1. The applied wing load at Frame 405 is sensitive to speed, and produces increasingly greater tension at the failure section as speed is increased with load factor kept constant. Increasing load factor reduces tensile load, so that maximum tensile stresses in the fatigue specimen are experienced at the inner flange at WL 89 at a speed of 340 knots at negative load factors.

2. The test spectrum randomized the fatigue loading conditions so that there are a large number of speed cycle changes. The only speed cycle changes considered in the original spectrum were ground-air-ground cycles, which are a small percentage of the speed cycles in the test spectrum. The large number of speed cycle changes caused the most fatigue damage.

3. There is some out-of-plan bending induced in the inner flange of the frame below the lower auxiliary longeron. This bending induced additional stresses of several thousand psi at the failure site, with the attendant decrease in fatigue life.

- At 1.02 lifetimes cracking was discovered on the fuselage skin panel forward of Sta 468. Subsequent inspection revealed numerous cracks in the radius of the panel beads. Cracks were also discovered at rivet holes attaching to the upper longerons and upper auxiliary longerons.

Strain surveys and subsequent investigation revealed the following facts and conclusions.

1. <u>Skin Panels</u>

   a. The cracking in the exterior skin panel assembly originated in the 7075-T6 beaded skin at the countersunk holes. This beaded pan skin is the inner sheet of the bonded assembly consisting of .040 inch 2024-T3 clad skin and the .050 inch 7075-T6 beaded pan.

b. The cracking was induced by high local bearing stresses in the approximately .020 inch thick cylindrical portion of the hole which resulted from countersinking .070 inch into a total metal thickness of .090 inch.

c. Contributing factors were: (1) a concentration of stresses around the fastener holes adjacent to the ends of the beads due to the redistribution of stresses around the beads, and (2) the stress cycles caused by speed changes.

d. The cracking was confined to this shear panel (LH and RH). The panel is unique in that it carries the highest shears and has deep countersinks.

e. The cracks had not reached a stage of rapid growth.

2. <u>Upper Auxiliary Longeron</u>

a. The cracking of the beads in the upper auxiliary longeron web and the exterior skin panel are unrelated except for the common speed and load factor spectrum.

b. The cracking of the beads was caused by bending stresses aggravated by a sharp radius in the underside of the bead.

c. The beads in the right hand side auxiliary longeron web had, in general, a less severe radius than those on the left hand side which accounted for the more general cracking found on the left hand side.

d. The stresses in the upper auxiliary longeron beaded web forward of frame 405 and aft of frame 468 are substantially lower than the panel where the cracking occurred. Assuming that the same stress concentration exists in these beads, analyses show them to have adequate life.

- Between 1.16 and 1.20 lifetimes, Hi-Lok fasteners failed outboard of WS 110 splice. Further inspection revealed a crack in the mid-spar web originating in an open tooling hole.

- Numerous front spar fastener failures were also found.

Subsequent test and investigations led to the following conclusions:

1. The failure of the mid-spar web was due to the absence of a plug rivet in the tooling hole.

2. The failures of the fasteners through the lower surface and front spar were due to oversize holes in the presence of load reversals.

- At 1.47 lifetimes, during a 25 percent inspection, the following discrepancies were noted:

  - A crack on the upper longeron plate, left hand side at the second screw hole forward of FS 468.

  - Several cracks on the door strap sill at FS 481.

  - Four small cracks at random fasteners at the door splice strap at FS 505.

  - One small crack on the door sill member at FS 444.

An investigation into the cause of these failures has led to the following conclusions:

1. The crack in the upper longeron plate, P/N 160D213001-49 was due to a badly scored hole. The scoring of the hole surface has been attributed to springing of the structure when the fatigue aircraft was reworked to remove the fuselage side skins between Stations 405 and 468. It appears that the damage was done while trying to force the parts to fit on reassembly. This condition is unique to the fatigue aircraft, in that none of the other aircraft have had the side skins removed.

2. In the initial investigation, it was found that the hole in question had less edge distance than the drawing requirement, .32 inch instead of .44 inch. A check of the stress concentration factors shows an increase of approximately 3 percent, and this is not considered significant. The fatigue analysis shows a life well in excess of 4 lifetimes, even with the smaller edge distance. The RH longeron in the area, where the holes have not been damaged, shows no signs of incipient fatigue cracks. The fuselages in the production line have been inspected, and all the holes in the longeron in this area have proper edge distance, and proper hole quality.

3. The failure of the frame 481 strap, P/N 160D212007-73, was due to loads introduced by the fuselage fuel tank access doors, acting as part of the primary fuselage bending structure. These doors were not intended to carry primary fuselage bending loads, but evidently the fit of the fasteners was close enough so that significant loads were carried. A strain survey in the fatigue aircraft confirmed this fact. The condition has been corrected in the fatigue aircraft by opening the holes for the fasteners by approximately 1/32 inch, from .192 inch to .221 inch for the No. 10 screws and from .252 inch to .281 inch for 1/4 screws. The strain survey was repeated with the larger holes, and showed that the door loads had been cut approximately in half, with negligible increase in the upper longeron stresses.

- Between 1.43 lifetimes and 1.82 lifetimes there were a few miscellaneous fastener failures primarily in the wing. These were considered isolated occurrences largely associated with improper fit holes.

- At 1.82 lifetimes a crack was discovered in the lower wing skin, left hand side, at approximately WS 118. The crack, which was six inches long, ran from the aft edge of the main skin, through a rivet hole, and forward through a cold-worked drain hole. Just prior to this failure (1.4 percent earlier) two fasteners in this area had cracked heads and were replaced, but no skin crack was observed. An investigation of this failure has established the following:

1. A metallurgical examination of the fracture surface has shown that the crack initiated at the aft side of the rivet hole, grew aft to the skin edge, and then forward through the drain hole.

2. There was no crack in the right hand wing skin, although an eddy current probe showed the existence of an anomaly in the bonded doubler at the rivet hole just inboard of the previously mentioned crack initiation point. Dye penetrant did not confirm the existence of a crack. This indication was on the forward side of the hole, whereas the maximum stress level is on the aft side of the hole.

3. Analysis has shown that there is a stress concentration induced by the rivet hole and by a change in wing skin width created by a tab designed to pick up the splice fitting at WS 110. The super position of both these effects increases the stress level at the aft side of the hole beyond the desired design level.

During a thorough inspection of the aircraft at two lifetimes, it was discovered that cracking had occurred in a number of areas. The following is a list of those areas:

- A small crack initiation site was found at the most inboard attachment in the upper line of attachments through the right hand fin mid-spar to horizontal stabilizer attachment fitting.

- Local skin cracks in the fuselage trough door at FS 483 right hand side.

- Cracks emanating from plate nut holes around the front spar web access cutout between WS 90 and WS 110 on the right and left hand wings.

- Cracks emanating from plate nut holes around the mid-spar web access cutout between WS 90 and WS 110 on the left hand wing.

- Cracking in the flap track on the outboard end of the outboard flap.

- Cracks emanating from fastener holes around the mid-spar web access cutouts between WS 23 and WS 44 and between WS 44 and WS 66 on the right hand wing.

An A-10 Structures Review Team (SRT) was convened to review the results of the two lifetimes of full scale fatigue testing, inspection and analyses. The SRT team concluded that the durability and damage tolerance has been adequately demonstrated for the required 6,000 hours of design service usage.

The SRT team also noted that an 8,000 hour life requirement was being used for the Air Force's other new fighter aircraft and recommended that the A-10 SPO determine the need for extending the present 6,000 hour life to 8,000 hours for the full scale fatigue test.

The results of the test extension are discussed in Reference (11) SA 160R9401.

<u>Service Problems</u>

The following A-10 service problems have been reported:

- Cracking of most forward nacelle ring frame (Prototype #1)

- Loosening of door fasteners near the gun muzzle (Prototype #1)

- Loss of a leading edge slat in flight (Prototype #1)

- Failure of a rudder lower attachment fitting (Prototype #1 and #2)

- Cracking of antenna shield on gun access door no. F-3 (DT&E #1)

- Buckling of wing/fuselage fillet, door no. F-26 (DT&E #1 and Static)

- Fuselage R/H Avionics door opened in flight, door no. F-10 (DT&E #3)
- Pilot access ladder door opened in flight (DT&E #3)
- Cracks were discovered in the fuselage fence (DT&E #2 and #5)
- Cracks were discovered in the wing/fuselage fillet, door numbers F-28 and F-89 (DT&E #2)
- Aileron vibration problems were discovered on ships 261 and DT&E #4
- Numerous incidences of cracks in the fuselage fence
- Failure of boarding ladder stops during ladder deployment (DT&E #2)
- Retention failure of the nacelle shroud door (Ships #24, #12, #38)
- Access panel fasteners pulling through
- Duct panel fasteners pulling through
- Buckling and cracking of bottom fuselage skin on a number of ships
- Cracking of leading edge skin under slat (Ships #8, #9)
- Rotation of the upper torque arm pin (Ships #7, #71)
- Cracked ribs supporting wing to fuselage fairing (Ships #8, #12)
- Malfunction of the single point refueling door latch (DT&E 1-6)

3.5.2 A-7D Aircraft

A review of the A-7D aircraft with OC-ALC personnel produced the following information pertaining to the durability of these aircraft.

- o Primary structure is good with major critical points being located on the wing.
- o Most A-7 structural problems have dealt with material design problems.
  - Material sections reduced to save weight resulted in overkill
- o Panels and doors are not interchangeable.
  - Too much hand-fit
- o Wheels are a major structural problem
  - 2024 forging - weak spoke design
- o Cracks occurring in .020 inch chem-milled skin
- o Corrosion problem in fin/tail section
- o Canopy rigging and latching problem
- o Maintenance and rigging costs are very high dollar items on the A-7D aircraft
- o Aircraft is not being used to its design capability because of engine problems; consequently the aircraft should last longer than its designed life (4000 hours).

o Other A-7 Data

The following datawere extracted from two reports published on the A-7 test program. These reports are

1. Fractographic Examination of A-7D ASIP Initial Quality Specimens - LTV Report No. 75-53452-078 (Reference 12).

2. An Equivalent Initial Flaw Analysis of the A-7A Wing Fatigue Test (Reference 13).

Tests were performed to evaluate the manufacturing quality of the A-7 aircraft. Test specimens were cut from a production A-7A aircraft lower wing skin which had been used as a gun fire target. This aircraft had approximately 700 hours so the probability of cracking was extremely low.

The specimens were cyclic tested to failure using a block test spectrum with each block consisting of 5000 cycles of 0-20 ksi followed by 0-30 ksi for 100 cycles. Fractographic analysis of each failure was then accomplished.

From the fractographic analysis,the following results and conclusions were made.

1. The straight shank holes usually began cracking in the bore of the hole near the surface whereas the counter-sunk holes most often began cracking at the inside radius of the small diameter portion of the hole.

2. Metallurgical investigation of the crack origins revealed two types of initial flaws were occurring; namely, pits in the holes and mechanical induced flaws from machining.

3. A survey of the cracks indicated six out of forty-four (13.6%) began from mechanical flaws while the remaining thirty-eight (86.4%) began from anodized pits. The material was 7075 aluminum.

4. The mechanical flaws were induced during drilling of the holes.

5. The pitting was induced by processing prior to anodizing or by the anodize process itself.

SECTION IV

OBSERVATIONS AND CONCLUSIONS

Structural durability data presented in Section III are evaluated and discussed in this section. Observations and overall conclusions are presented. Essential Conclusions of the structural durability survey are:

1. The most frequent structural durability problems occurring in the in-service Air Force aircraft surveyed, in order of occurrence, are: cracking, corrosion and fastener-related problems.

2. Repair/maintenance cost data were not available for assessing durability-related problems for the aircraft surveyed.

3. A uniform format is needed for documenting durability-related problems at the Air Logistic Centers. Data should be compiled in useful formats for storage and retrieval and should be periodically updated.

Further details of the data evaluations are presented and results are discussed in this section.

A Structural Assessment Form (Appendix A) was devised for documenting in-service data. This form was later discarded due to the type, form and quantity of information available at the Air Logistic Centers visited. The Structural Assessment Form was time consuming to fill out. Although this form was discarded, it would be useful for documenting in-service inspections or teardowns. Several information sources were used for the Structural Assessment: Analytical Condition Inspection (ACI) reports, Aircraft Structural Integrity Plan (ASIP), direct contacts with ACI personnel, and contractor reports. Since the ACI reports contained the most quantitative information, these reports were used extensively.

There are numerous ACI reports for various aircraft systems. However, the ACI results for each aircraft were not reported in the same format for direct comparisons. For example, one ACI may list corrosion and stress-corrosion occurrences individually and another ACI may only cite occurrences of "corrosion." Another example is based on the C-5A transport. ACI reports for CY 76 and CY 77 compile occurrences and general location of fatigue cracks but fail to

connect any of those cracks with fastener holes. Structural assessment data for different aircraft were cataloged as follows so that results could be combined for purposes of assessing general trends:

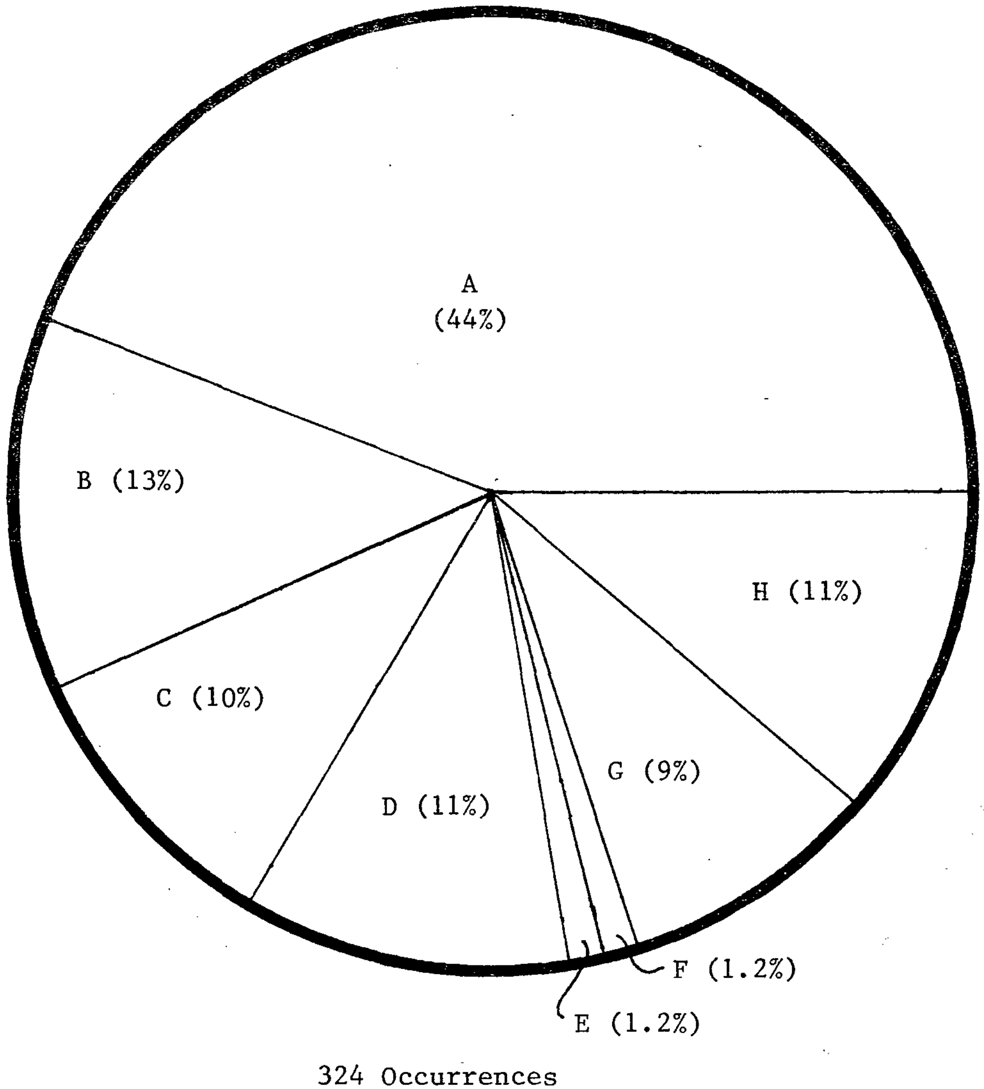
A. Cracking: Holes, plates, radii, fittings, etc.
B. Corrosion: Stress and/or any other
C. Fastener Related: lose, missing, failed, etc.
D. Dents/Nicks/Scratches
E. Honeycomb Delamination/Damage
F. Fastener Hole Related: out-of-round, etc.
G. Wear: Chaffing, Fretting, etc.
H. Maintenance: Improper or faulty practices
I. Misc: Specific system particularities

These categories are somewhat ambiguous but do allow general comparisons of the various aircraft systems.

A summary of pertinent ACI results is shown in Figures 51 through 57, for the T-39A trainer, F-4/C/D/E, RF-4C fighter, F/FB-111A fighter-bomber, F-111C bomber, C-5A transport, and F-15A fighter, respectively. These aircraft systems have been in the Air Force's inventory for varying lengths of time. The approximate times at which the various systems went into service are as follows:

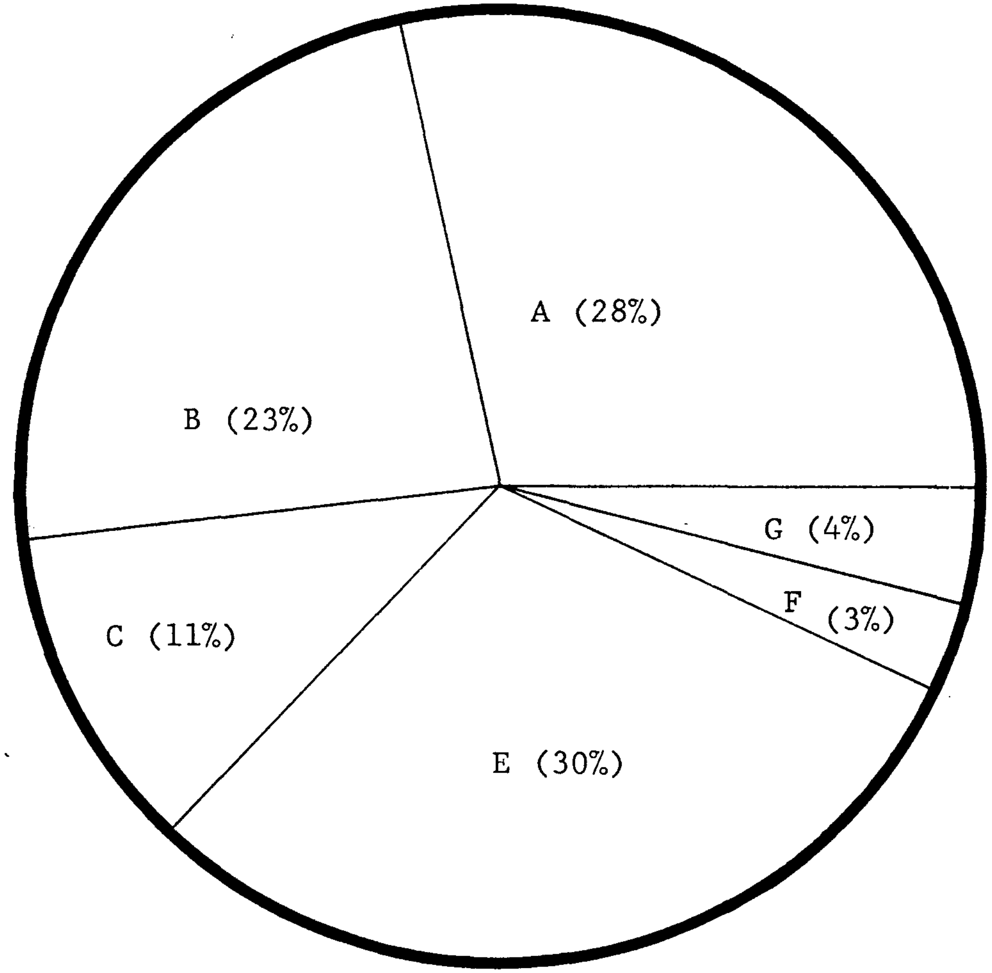
T-39A	Early 60's
F/RF-4	Early 60's
F/FB-111A	Mid 60's
F-111C	Early 70's
C-5A	Late 60's to Early 70's
F-15A	Early 75

The T-39A trainer, Figure 51 , provided some very interesting results. This particular ACI was very comprehensive, perhaps due to the age of this system. Obviously cracking was the predominate occurrence, but of special interest was the number of maintenance-related incidents. The incidents, which are not mentioned in all ACIs, are the results of carless induced maintenance. Those incidents include abuse of "no-step" areas or dropping of tools on sensitive areas. Also included are incidents of improperly performed maintenance, such as double or misdrilled holes, and other damage associated with the drilling of holes. It should be noted that although the amount of cracking shown may seem excessive, when considered relative to the time in service this number may approach mean values.



INCIDENT		CUMMULATIVE OCCURRENCES
A	Cracking	142
B	Corrosion	41
C	Fastener Related	31
D	Dents/Nicks/Scratches	36
E	Honeycomb Damage/Delamination	4
F	Fastener Hole Related	4
G	Wear	29
H	Maintenance Related	37
	Total	324

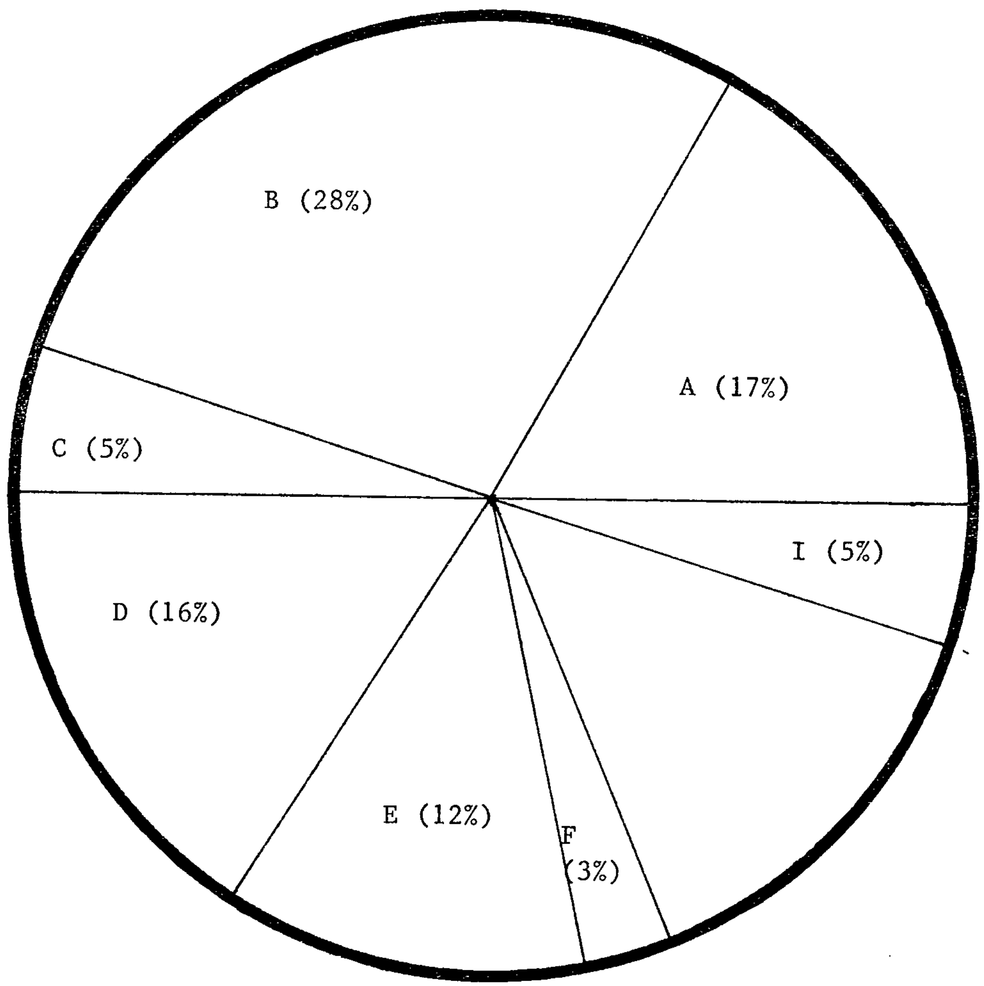
Figure 51 T-39A Analytical Condition Inspection (3 Aircraft Surveyed)



181 OCCURRENCES

INCIDENT		CUMULATIVE OCCURRENCES
A.	Cracking	51
B.	Corrosion	42
C.	Fastener Related	19
D.	Dents/ Nicks/ Scratches	0
E.	Honeycomb Damage/ Delamination	55
F.	Fastener Hole Related	6
G.	Wear	8
H.	Maintenance	0
I.	Miscellaneous	0
	Total	181

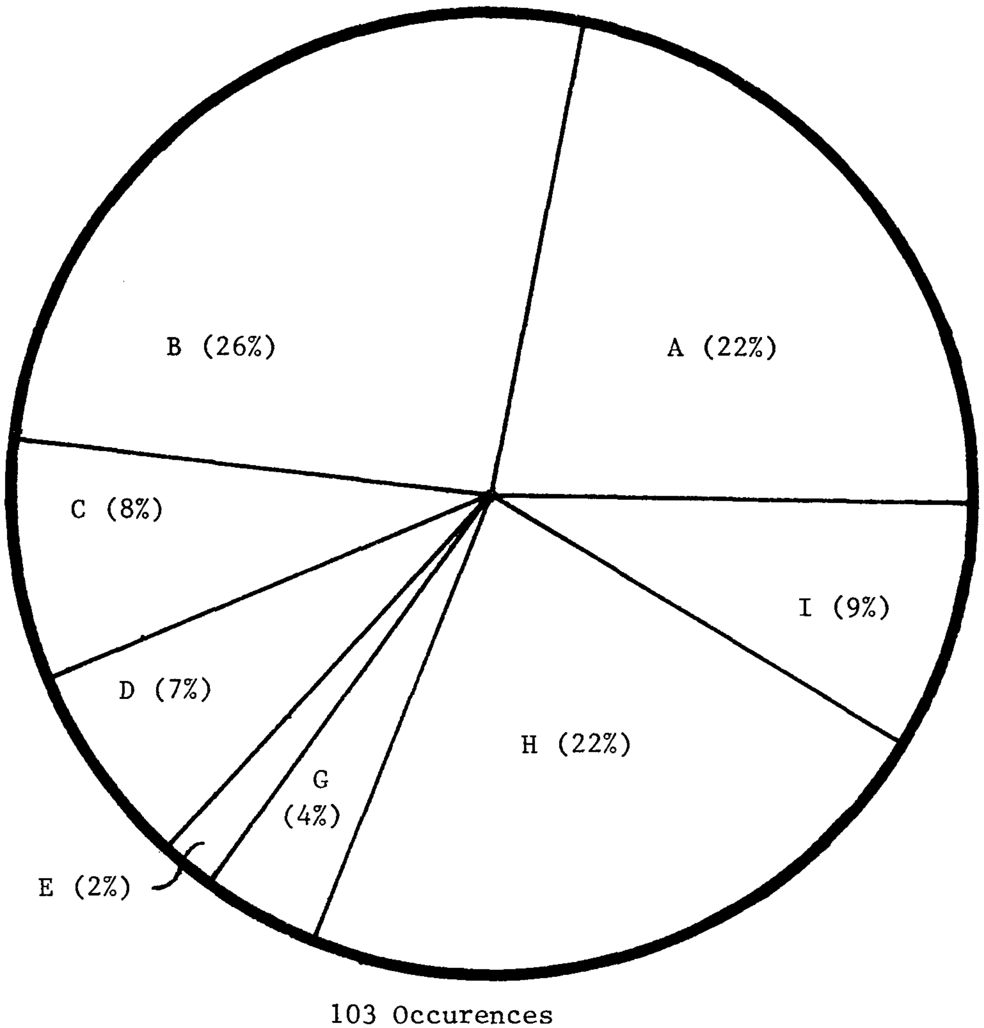
Figure 52 F-4C/D/E and RF-4C Analytical Condition Inspection (44 Aircraft Surveyed).



	INCIDENT	CUMULATIVE OCCURRENCES
A	Cracking	11
B	Corrosion	18
C	Fastener Related	3
D	Dents/Nicks/Scratches	10
E	Honeycomb Damage/ Delaminations	8
F	Fastener Hole Related	2
G	Wear	9
H	Maintenance	0
I	Miscellaneous	3
	Total	64

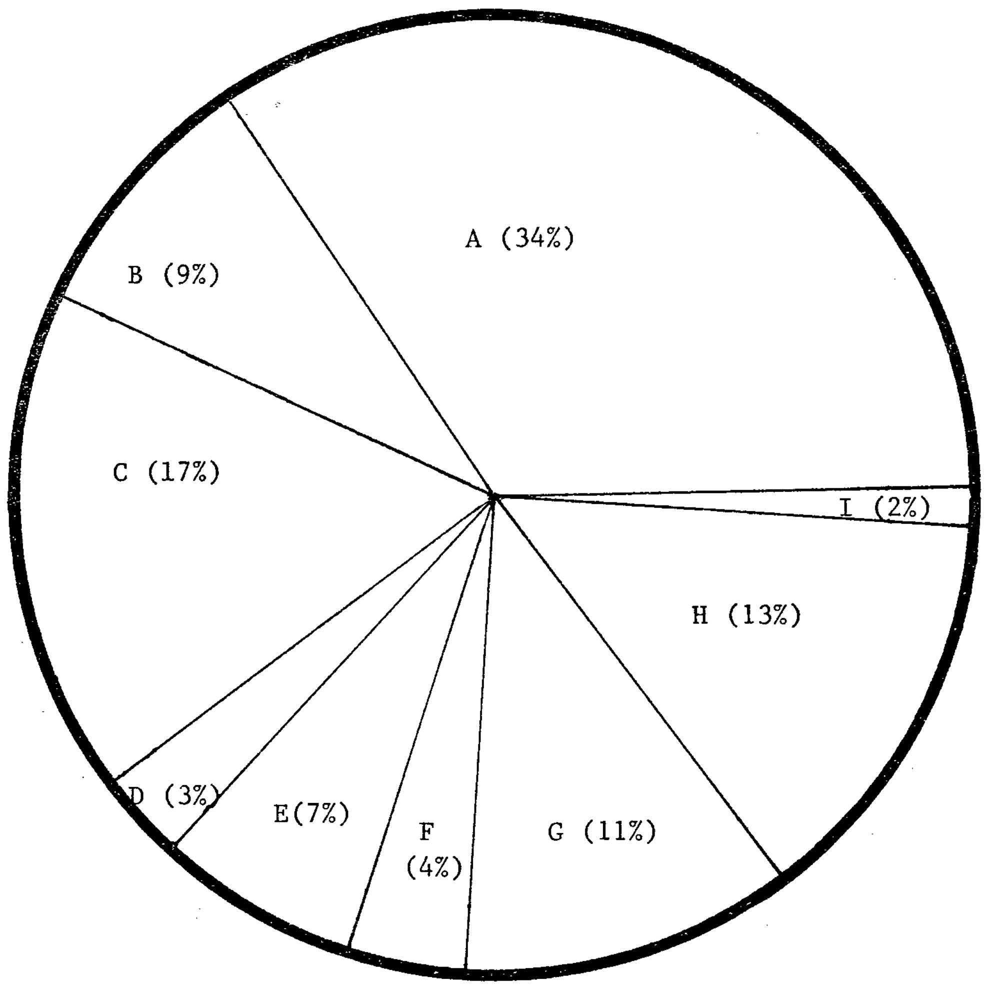
64 Occurrences

Figure 53 F/FB-111A Airframe Analytical Condition Inspection



	INCIDENT	CUMULATIVE OCCURRENCES
A	Cracking	23
B	Corrosion	27
C	Fastener Related	8
D	Dents/Nicks/Scratches	7
E	Honeycomb Damage/ Delamination	2
F	Fastener Hole Related	0
G	Wear	4
H	Maintenance	23
I	Miscellaneous	9
	Total	103

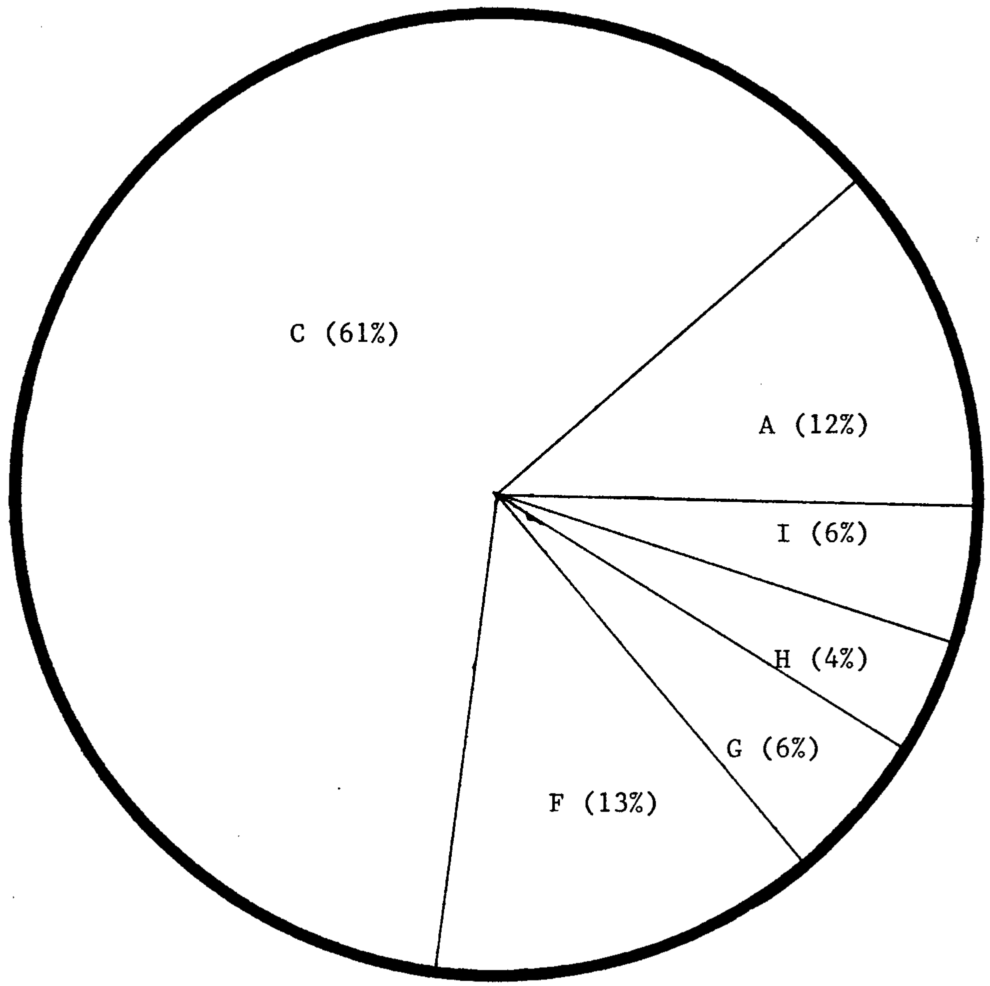
Figure 54 F-111C Analytical Condftion Inspection (4 Aircraft Surveyed)



289 Occurrences

INCIDENT		CUMULATIVE OCCURRENCES
A	Cracking	98
B	Corrosion	25
C	Fastener Related	49
D	Dents/Nicks/Scratches	8
E	Honeycomb Damage/ Delamination	21
F	Fastener Hole Related	12
G	Wear	32
H	Maintenance	38
I	Miscellaneous, Pressure Seals, Etc.	6
	Total	289

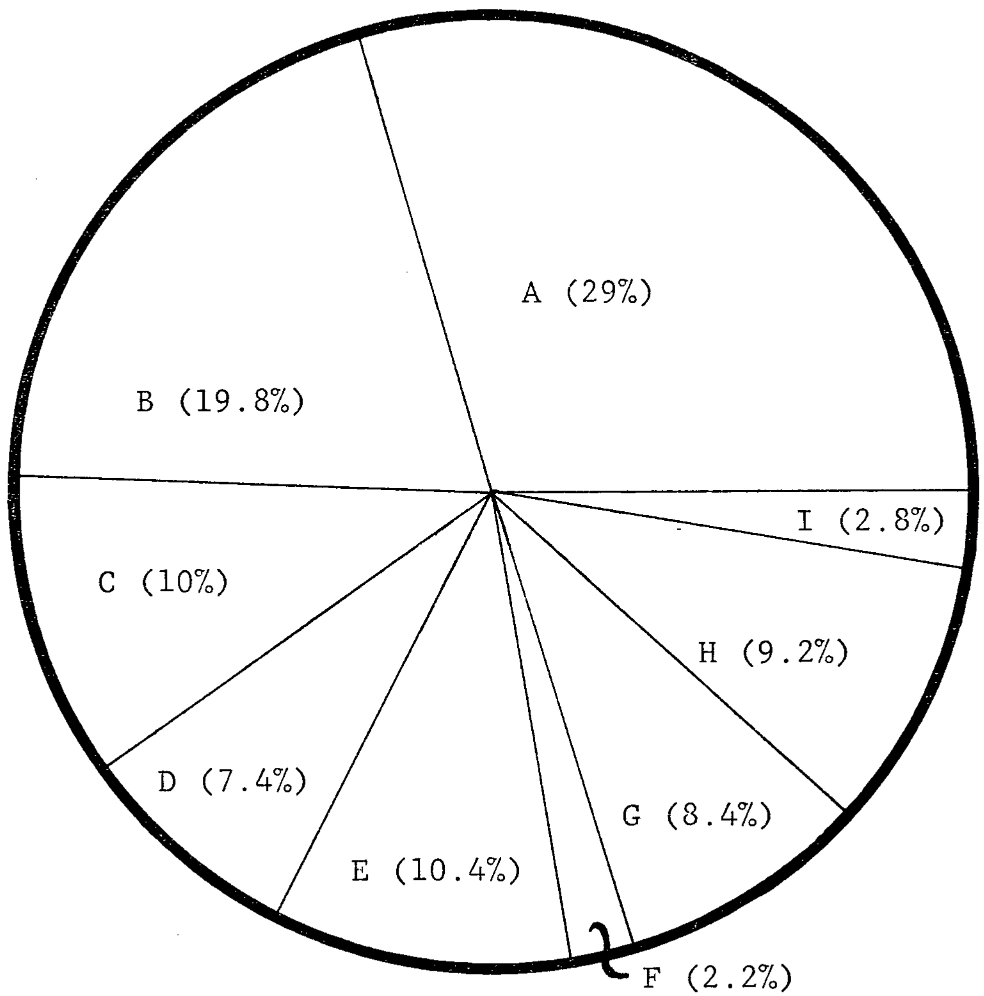
Figure 55 C-5A Analytical Condition Inspection Cumulative Occurrences for CY76 and CY77



86 Occurrences

INCIDENT		CUMULATIVE OCCURRENCES
A	Cracking	10
B	Corrosion	0
C	Fastener Related	53
D	Dents/Scratches/Nicks	0
E	Honeycomb Damage/ Delamination	0
F	Fastener Hole Related	11
G	Wear	4
H	Maintenance/Manufacturing	3
I	Miscellaneous	5
	Total	86

Figure 56 F-15A Analytical Condition Inspection



INCIDENT		% CUMULATIVE OCCURRENCE
A.	Cracking	29
B.	Corrosion	19.8
C.	Fastener Related	10
D.	Dents/Nicks/ Scratches	7.4
E.	Honeycomb Damage	10.4
F.	Fastener Hole Related	2.2
G.	Wear	8.4
H.	Maintenance	9.2
I.	Miscellaneous	2.8

Figure 57 Average Cumulative Distribution by Percentage of Incidents from ACI for T-39A, F-4C/D/E, RF-4C, F-111, F-111C, and C-5A Systems.

The ACI summary for the F-4C/D/E and RF-4C is shown in Figure 52. This ACI was conducted during CY77 and represents a survey of 44 aircraft. These aircraft have seen lengths of service time that vary from one to fourteen years. The mean service lives are listed below:

F-4C	12.8 Yr
F-4D	10.7 Yr
F-4E	6.9 Yr
RF-4C	10.0 Yr

This group of aircraft is representative of a mid-60's system implementation.

As shown in Figure 52 there are several incidents worthy of note. Occurrences of cracking, corrosion, fastener related incidents, fastener-hole-related incidents, and wear are relatively normal. On the other hand the amount of honeycomb damage is abnormally high. There are no reported occurrences of maintenance-related type incidents, i.e., dents/nicks/ scratches, maintenance, or miscellaneous incidents.

Figures 53 and 54 are ACI summaries for F/FB-111A and F-111C aircraft, respectively, Although these aircraft, which are quite similar, have seen different service times, they show similar trends. The predominant examples are in cracking, corrosion, and in fastener-related incidents. The F/FB-111A ACI reports no maintenance-related problems, whereas F-111C shows 22% of problems maintenance related.

Analytical Condition Inspection results for the C-5A transports are shown in Figure 55. Of interest here are the number of fastener-related and honeycomb-related problems. The high occurrence of honeycomb delaminations is to be expected since aircraft of this type use, on a percentage basis, more structural honeycomb.

The F-15A ACI results, shown in Figure 56 do not reflect any trends, to be established later, seen in the aforementioned systems. This particular aircraft, the youngest fully deployed system in the Air Force inventory, is representative of state-of-the-art design and materials technology. The combination of age and technology yields the low or non-existant percentages in all categories other than fastener related. This particular ACI has a limited data base and therefore is not included in the following trends.

The results shown in Figures 51 through 56 are combined into one composite figure, as shown in Figure 57. The average frequency of occurrence for a given incident during any inspection are noted. The numerical ranking of incidents is as follows:

1. Cracking
2. Corrosion
3. Maintenance
4. Fastener Related
5. Dents/Nicks/Scratches
6. Wear
7. Miscellaneous
8. Honeycomb Delaminates/Damage
9. Fastener Hole Related

The above rankings are considered to be "generally" applicable to all aircraft systems in the Air Force inventory. Granted each system undergoes usage variations and has its own particular idiosyncrasies that surface in ACI type reports. But these trends are believed to be consistent. The above ranking may change once service data are obtained from some of the newer aircraft e.g., F-15, F-16, F-18 etc.

Several interesting correlations can be made to demonstrate the consistency of the ACI findings. Table 22 shows the percent of occurrence for the T-39A, F/FB-111A, F-111C, C-5A, and F-4 aircraft. In Table 22, consistent trends are noted for the combined percent occurrences of cracking (A) and corrosion (B) for the various aircraft surveyed. For example, the % occurrences for A + B ranges from 43% to 57%. In the same manner the combination of fastener related incidents, dents/nicks/scratches, and fastener-hole related incidents yield less consistent values. If the miscellaneous category is included, a nearly constant value is obtained. Other combinations yielding uniform values are the combination of "maintenance"-related categories, i.e., dents/nicks/scratches and fastener-hole related and maintenance-related incidents.

In most of these combinations there is no interrelationship. The point of these examples is to show that for all aircraft systems of some minimum age a relatively constant percentage of each maintenance exercise will address the repair of certain incidents. The conclusions that can be drawn from these data are:

1. For this assessment, the Analytical Condition Inspection was the best source of data for evaluating structural durability.

TABLE 22 Percent Occurrences For ACl Recorded Incidents

INCIDENT	ACI % OCCURRENCES					
	T-39A	F-4	F/FB-111A	F-111C	C-5A	AVE
A. Cracking	44	28	17	22	34	29
B. Corrosion	13	23	28	26	9	19.8
C. Fastener Related	10	11	5	8	17	10
D. Dents/Nicks Scratches	11	0	16	7	3	7.4
E. Honeycomb Damage	1.2	30	12	2	7	10.4
F. Fastener Hole Related	1.2	3	3	0	4	2.2
G. Wear	9	4	14	4	11	8.4
H. Maintenance	11	0	0	22	13	9.2
I. Miscellaneous	0	0	3	9	2	2.8
A + B	57	51	45	48	43	48.8
C + D + F	22.2	14	24	15	24	19.8
C + D + F + I	22.2	14	27	24	26	22.6
D + H + F	23.2	3	19	29	20	18.8

2. In general, similar types of structural problems occur for different aircraft.

3. As the rate of corrosion occurrence decreases with improved materials technology, other material-related problems occur, yielding a relatively constant percentage of "structural occurrences."

4. The F-15A fighter exhibits anomalous behavior relative to other aircraft systems, possibly establishing a trend of increased structural durability.

5. Improved methods are needed for tracking structural incidents at the maintenance depot level to allow a more accurate evaluation of structural performance.

6. A voluminous data bank for durability was collected during the survey for most of the aircraft in the U.S. Air Force active inventory. Detailed assessment of these data is beyond the task, funds, and schedule for this report. These data are on file at the contractor's facility.

APPENDIX A

DURABILITY METHODS DEVELOPMENT

STRUCTURAL ASSESSMENT RESULTS

FLEET INFORMATION: Fill in blanks and/or check (√) applicability.

A/C DESIGNATION: ______________________________
Fighter ( ); Trainer ( ); Bomber ( ); Cargo/Transport ( );
Other ( )

SYSTEM MGT: SMALC ( ); OOALC ( ); SAALC ( ); WRALC ( ); OCALC ( )

DATE TRANSFERRED TO AFLC: ______________________________

PRIMARY BASE: ______________________________

PRIMARY MISSION TYPE: ______________________________

HOW IS AIRCRAFT BEING USED? e.g. MORE/LESS SEVERE THAN DESIGN USAGE:

____

REMARKS: ______________________________

FORM OF STRUCTURAL AND/OR MATERIAL DEGRADATION

MODE:

( ) STATIC
( ) FATIGUE CRACKING
( ) CORROSION FATIGUE
( ) FRETTING
( ) WEAR
( ) GALLING
( ) OTHER
REMARKS ______________________________
____
____

STRUCTURAL DEGRADATION DOCUMENTATION

____
METALLURGICAL OR FAILURE ANALYSIS REPORT NO. AND TITLE

____
DISCREPANCY DATA SHEET, LOG ENTRY, ETC; TO FULLY DOCUMENT FINDINGS

PART NO. AND TITLE ______________________________

____

AIRCRAFT LOCATION: ( ) WING
( ) FUSELAGE
( ) VERT. TAIL
( ) HORIZ. TAIL
( ) CONTROL SURF
( ) SPEED BRAKE
( ) OTHER
LOC. DETAIL ______________
e.g. station,

____
B.L., W.L., outer skin, inner skin, etc.

____

LOCATION OF ORIGIN: ( ) FASTENER HOLE
( ) RADII
( ) NOTCH
( ) COUNTERSINK/
C'BORE
( ) OTHER
REMARKS ______________________

____

CRACKING DESCRIPTION: ______________________________

____
(a) origin, (b) initial flaw size, (c) crack length, (d) other.

MATERIAL: ______________________________
Form, temper, heat treat, etc.

DESIGN: ( ) ORIGINAL: ( ) MODIFICATION

DISCOVERED, How, When: ( ) ROUTINE INSP.
( ) PHASE INSP.
( ) OTHER
REMARKS ______________________

____

____

PREVIOUS CRACKING/DEGRADATION EXPERIENCE:

( ) INITIAL
( ) REPEATED
( ) FREQUENCY
REMARKS ______________________________

PROBABLE CAUSE

DESIGN RELATED (   )

( ) SHARP RADII
( ) NOTCHES
( ) SEALANT GROOVES
( ) FASTENER RELATED (TENSION BOLTED JOINTS)

( ) COUNTERSINK/C'BORE
( ) SHORT EDGE DISTANCE
( ) OTHER
REMARKS ______________________
____

( ) IS THIS INFO. DOCUMENTED IN AFOREMENTIONED REPORT (i.e., METALLURGICAL, FAILURE ANALYSIS, ETC.)

MATERIAL RELATED (   )

( ) PLATE, ( ) FORGING, ( ) CASTING, ( ) EXTRUSION
( ) IMPROPER SELECTION
( ) DEFECTS
FLAWS: ( ) METALLURGICAL, ( ) MANUFACTURING, ( ) OTHER
DENTS: ( ) TOOL MARKS, ( ) HANDLING, ( ) OTHER
SCRATCHES: ( ) AXIAL, ( ) CIRCUMFERIENTIAL
( ) IMPROPER HEAT TREATMENT
( ) CHEMISTRY
( ) OTHER
REMARKS ______________________

SURFACE TREATMENT (   )

( ) CHROMIC ANODIZED
( ) SULFURIC ANODIZED
( ) CHEM. FILM
( ) FINISH: ______ RMS
( ) SHOT PEENED

( ) OTHER
REMARKS ______________________
____
____
____

HOLES (   )

( ) WEAR AS IN PANEL REMOVAL
( ) DRILL DAMAGE
( ) ELLIPTICAL
( ) COUNTERSINK
( ) DOUBLE DRILLED
REMARKS ______________________

MANUFACTURING RELATED (   )

( ) MACHINING
( ) FORMING
( ) ETCHING
( ) DIMENSIONAL ACCURACY
( ) IMPROPER TOOLING
( ) OTHER
REMARKS ____________________
____

BONDING RELATED (   )

( ) MATERIALS
( ) ADHESIVES
( ) PROCEDURES
( ) OTHER
REMARKS ____________________

ASSEMBLY/INSTALLATION RELATED (   )

( ) PRE-LOAD
( ) MISMATCH/MATING
( ) MOD/REPAIR
( ) OTHER
REMARKS ____________________

ENVIRONMENT RELATED (   )

( ) SUSTAINED STRESSES
( ) CORROSIVE ENVIRONMENT
( ) GALVANIC CELL
( ) PITS, SCRATCHES
( ) OTHER
REMARKS ____________________

MAINTENANCE RELATED (   )

( ) IMPROPER HANDLING
( ) USAGE FREQUENCY
( ) INADEQUATE INSPECTION
( ) IMPROPER RIGGING
( ) SERVICE INDUCED
( ) OTHER
REMARKS ____________________

ARE COST DATA AVAILABLE FOR REPAIR AND/OR MOD OF THIS DISCREPANCY/ FAILURE? YES ________ NO ________

COST DATA ____________________

LIST OF ANALYTICAL CONDITION INSPECTION (ACI) REPORTS ____________________

APPENDIX B

T-38 TEARDOWN INSPECTION RESULTS

A. BACKGROUND

The CY 76 T-38 ACI was a teardown inspection of ten Tactical Air Command wings that were retired at service life. The findings of the ACI indicated the presence of generalized fatigue cracking in critical area fastener/drain holes. The fatigue condition of high time Air Training Command (ATC) wings is unknown, and present NDI techniques are considered incapable of detecting the extremely small crack sizes expected. Due to the susceptibility of the lower wing skin to fatigue cracking and the advancing age of the fleet, it is essential to determine the condition of high time ATC wings to implement long range logistics support of the ATC fleet.

B. PURPOSE

The purpose of this program is to provide special teardown inspection data on the condition of three (3) high time ATC wing assemblies, emphasizing fatigue cracking in the fastener/drain holes, corrosion effects, and mechanical condition of the skin regarding the presence of rework areas and mechanical damage which may create stress raisers in the assembly.

C. SCOPE

The scope of the program was to perform a special teardown inspection on three (3) high time ATC wing assemblies. The designated area inspected was the lower wing skin spanwise between the landing gear ribs and fore/aft from the 39% spar to the 44% spar, and all speed brake attach and drain holes. The internal components of the wing assembly in this area were also inspected. The levels of inspection utilized in this program included visual, fluorescent penetrant, fractographic, and SEM inspection of selected specimen flaws. The inspected high time wing assembly serial numbers and associated flight time hours are listed below:

- Wing No. 11 - S/N 63-8202, 6257.6 hours
- Wing No. 12 - S/N 65-10440, 5445 hours
- Wing No. 13 - S/N 62-3618, 8277 hours

D. TEARDOWN INSPECTION FINDINGS

1. Wing Assembly No. 11, S/N 63-8202

a. Visual Inspection. Initial visual inspection of this wing assembly revealed the presence of an epoxied scab patch at the 44% spar which covered hole numbers 11-H-1345 through 11-I-1363. The scab patch was not painted nor was the wing skin surface in the vicinity of the patch. Once the fasteners were removed from this area closer inspection revealed:

- the presence of a shim between the scab patch and the skin covering hole numbers 11-H-1353 and 11-H-1354. This shim was installed to cover the rework area over these holes having a radius of 1.1 inches, ground to a depth of .045 inch.
- seventeen (17) of the nineteen (19) countersinks in the scab patch were poorly drilled as evidenced by the gouges and thin flaking material at the countersink/epoxy interface.
- the scab patch countersinks were not drilled for oversized jo-bolts as evidenced by the large bearing area of epoxy in the countersinks (refer to Figure 3 for an illustration of a typical scab patch fastener hole)

Thirty-two (32) fastener holes were reported to have gouged or scored hole/countersink surfaces and out-of-round shape. In addition to these flaws, the fastener holes listed below contained the flaws noted, for which photographs are presented in Appendix A.

- 11-G-1118: Intergranular corrosion-corrosion penetration at countersink
- 11-H-1353: Misdrill or gouge in hole
- 11-I-1356:
  11-I-1357: Intergranular corrosion between the two fastener holes - corrosion penetration at score in countersink of hole no. 11-H-1356

- 11-J-1948: Large delamination crack
- 11-K-2314: Metal smears in hole surface
- 11-M-1391: Possible stress corrosion penetration in countersink surface
- 11-R-1437:
  11-R-1438: Intergranular corrosion between the two fastener holes - corrosion penetration suspected at score in countersink of hole no. 11-R-1437
- 11-S-1245: Intergranular corrosion-corrosion penetration at countersink

Note that the intergranular corrosion flaws in hole nos. 11-I-1356, 1357 and 11-R-1437, 1438 are in approximately symmetrical locations along the 44% spar.

b. Fluorescent Penetrant Inspection. Table B-1 summarizes the fluorescent penetrant inspection flaws detected in the internal components of wing assembly No. 11. All of these reported flaws were within the acceptable limits specified by TO 1T-38A-3, Structural Repair.

c. Microscopic/Fractographic Inspection. Table B-2 summarizes the significant microscopic/fractographic inspection results. Of the twenty (20) specimens selected for SEM analysis, seventeen (17) contained scored countersink/hole surfaces which may have acted as stress raisers in the propagation of the fatigue or delamination cracks (refer to Appendix A for the data tables and SEM photographs).

TABLE B-1

SUMMARY OF FLUORESCENT PENETRANT INSPECTION, WING ASSEMBLY S/N 63-8202

Wing Assembly Component	Flaw Description	Length (in.)	Flaw Initiation
Lower Spar Cap, 39% Spar, Wing Cut Section H	Stress corrosion crack propagating from one (1) fastener hole	.09	Corrosion pitting
Lower Spar Cap, 44% Spar, Wing Cut Section I	Stress corrosion crack propagating from one (1) fastener hole	.27	Corrosion pitting
Upper Spar Cap, 39% Spar, Wing Cut Section P	Stress corrosion crack propagating through the web/flange radius	1.11	Corrosion pitting
Lower Spar Cap, WCS, FS 356, Wing Cut Section P	Stress corrosion crack propagating across one (1) fastener hole	.45	Corrosion pitting
Lower Spar Cap, 44% Spar, Wing Cut Section Q	Two (2) stress corrosion cracks propagating from one (1) fastener hole	.45 .42	Corrosion pitting
Lower Spar Cap, 44% Spar, Wing Cut Section R	Stress corrosion crack propagating from one (1) fastener hole	.40	Corrosion pitting

TABLE B-2

SUMMARY OF WING SKIN HOLE INSPECTION, WING ASSEMBLY S/N 63-8202

Total Inspected	Total Fatigue Cracks >.010 in.	Total Fatigue Cracks <.010 in.	Maximum Fatigue Crack Length (in.)	Total Dimensionable Delamination Flaws	Total Multiple Fatigue Cracks	Total Corrosion Flaws
419	1	66	.011 (11-I-1145)	4	4	261

2. Wing Assembly No. 12, S/N 65-10440

a. Visual Inspection. Forty-three fastener holes were reported to have gouged or scored hole/countersink surfaces and out-of-round shape. In addition, the flaws noted below were detected on the lower wing skin:

- Wing Skin Radius at 44% Spar, WS36, Wing Cut Section I: Additional radius machined in landing gear door lands. Additional fastener hole. Landing gear door fretting to a depth of .027 inch. Tang fretting.

- 12-P-1418: Fretting damage observed on the faying surface between the skin and root rib lower flange.

- Wing Skin Radius at 44% Spar, WS26, Wing Cut Section P: Multiple rework radii.

b. Fluorescent Penetrant Inspection. No flaws were detected on the lower wing skin or internal components of wing assembly No. 12 using fluorescent penetrant.

c. Microscopic/Fractographic Inspection. Table B-3 summarizes the significant findings of the microscopic/fractographic inspection. Of the 12 specimens selected for SEM analysis, all contained scored countersink/hole surfaces which may have acted as stress raisers in the propagation of the fatigue or delamination cracks.

TABLE B-3

SUMMARY OF WING SKIN HOLE INSPECTION, WING ASSEMBLY S/N 65-10440

Total Inspected	Total Fatigue Cracks >.010 in.	Total Fatigue Cracks <.010 in.	Maximum Fatigue Crack Length (in.)	Total Dimensionable Delamination Flaws	Total Multiple Fatigue Cracks	Total Corrosion Flaws
401	0	8	.004 (12-I-1360)	7	1	341

3. <u>Wing Assembly No. 13, S/N 62-3618</u>

a. <u>Visual Inspection.</u> Thirty-five fastener holes were reported to have gouged or scored hole/countersink surfaces and out-of-round shape. In addition, the flaws noted below were detected on the lower wing skin:

- 13-J-1371: Grind marks on skin surface around countersink. Possible corrosion in countersink.
- 13-K-2004: Metal smear in fastener hole.
- 13-K-2006: Metal smear in fastener hole.
- 13-M-2014: Intergranular corrosion at faying surface of hole.
- 13-P-2090: Intergranular corrosion at faying surface of hole.
- 39% Spar, Wing Cut Section J: Corrosion and fretting indicated the presence of foreign material between wing skin and lower spar cap.
- 44% Spar, Wing Cut Section J: Corrosion and fretting indicated the presence of foreign material between root rib and lower spar cap.
- Wing Skin Radius at 44% Spar, WS 26, Wing Cut Section J: Corrosion and fretting indicated the presence of foreign material between the wing skin and lower flange of the root rib.
- Root Rib, Wing Cut Section J: Corrosion and fretting indicated the presence of foreign material between the wing skin and the lower flange of the root rib.
- Wing Skin Radius at 44% Spar, WS36, Wing Cut Section J: Landing gear door fretting. Tang fretting.
- Wing Skin Radius at 44% Spar, WS 36, Wing Cut Section Q: Landing gear door fretting. Tang fretting.
- Wing Skin Radius at 44% Spar, WS 64.8, Wing Cut Section S: Rework Radius.

b. Fluorescent Penetrant Inspection. Table B-4 summarizes the fluorescent penetrant inspection flaws detected in the internal components of wing assembly No. 13. These flaws were within the acceptable limits specified by TO 1T-38A-3, Structural Repair.

c. Microscopic/Fractographic Inspection. Table B-5 summarizes the significant findings of the microscopic/fractographic inspection. Of the 21 specimens selected for SEM analysis, all contained scored countersink/hole surfaces which may have acted as stress raisers in the propagation of fatigue or delamination cracks.

TABLE B-4

SUMMARY OF FLUORESCENT PENETRANT INSPECTION, WING ASSEMBLY S/N 62-3618

Wing Assembly Component	Flaw Description	Length (in.)	Flaw Initiation
Lower Spar Cap, 44% Spar where it attached to root rib, Wing Cut Section P	Stress corrosion crack propagating through web/flange radius	1.79	Possible corrosion in web/flange radius
Upper Spar Cap, 44% Spar where it attached to root rib, Wing Cut Section P	One (1) stress corrosion crack propagating between two (2) fastener holes, one (1) stress corrosion crack propagating from one (1) fastener hole to the end of the flange	.76 .59	Corrosion pitting

TABLE B-5

SUMMARY OF WING SKIN HOLE INSPECTION, WING ASSEMBLY S/N 62-3618

Total Inspected	Total Fatigue Cracks >.010 in.	Total Fatigue Cracks <.010 in.	Maximum Fatigue Crack Length (in.)	Total Dimensionable Delamination Flaws	Total Multiple Fatigue Cracks	Total Corrosion Flaws
403	3	134	.017 (13-J-1939)	7	7	325

E. SUMMARY OF SIGNIFICANT FINDINGS

1. VISUAL INSPECTION

Of the 1223 holes inspected on the high time ATC wing assemblies, 110 holes were reported to have damaged countersink/hole surfaces and out-of-round shape. This condition was further supported by SEM photographs of selected hole specimens.

A significant finding of the visual inspection was the detection of obvious intergranular corrosion in fastener holes on wing assembly No. 11. In two instances (hole Nos. 11-I-1356-57 and hole Nos. 11-R-1437-38) the intergranular corrosion propagated between the fastener holes via corrosion entry at scores in the countersinks. These flaws were in approximately symmetrical locations at the 44% spar plane on the wing assembly, which indicates the susceptibility of this area to stress and intergranular corrosion flaws if not detected in periodic inspections. A third significant intergranular corrosion flaw (hole No. 11-S-1245) propagated from a score in the countersink to the wing skin surface where the skin material began to bubble and flake.

An additional significant finding of the visual inspection on wing assembly No. 13 was the indication of foreign material between the wing skin faying surface and the internal components. This indication was evidenced by the presence of fretted areas which contained corrosion effects. This condition if persistent could create stress raisers which could lead to stress corrosion cracking in the internal components.

2. FLUORESCENT PENETRANT INSPECTION

The fluorescent penetrant inspection did not reveal any significant findings since all detected cracks were within the acceptable limits. A recurring flaw, reported in the previous ACI, was detected in the 44% spar where it attaches to the root rib on wing assembly No. 13.

3. MICROSCOPIC/FRACTOGRAPHIC INSPECTION

Table B-6 summarizes significant crack data recorded from the microscopic/fractographic inspection. A significant finding of the SEM analysis is that 50 of the 53 specimens analyzed contained scores or gouges, and that fatigue cracks, where noted, appeared to propagate from these gouges.

TABLE B-6

SUMMARY OF FRACTOGRAPHIC INSPECTION

Wing Assembly S/N	Total Inspected	Total Fatigue Crack Length ≥.001 <.005 in.	Total Fatigue Crack Length ≥ .005 < .010 in.	Total Fatigue Crack Length ≥ .010 in.	Maximum Fatigue Crack Length (in.)
63-8202	419	52	14	1	.011 (11-I-1145)
65-10440	401	12	0	0	.004 (12-I-1360)
62-3618	403	112	35	3	.017 (13-J-1939)

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